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# Research Laboratories

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EAST HARTFORD, CONNECTICUT

NASA CR-53472;

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Study of the Application of
Electric Propulsion to Space Missions
Quarterly Progress Report No. 3
Contract ~~No~~. NASw-737

UNCLASSIFIED

REPORTED BY W. R. Fimple
W. R. Fimple Mar. 1964 38p refs

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J. L. Cooley
Chief, Evaluation Section

DATE March 1964

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Report B-910076-3

## Study of the Application of Electric Propulsion to Space Missions

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### SUMMARY

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Contract NASw-737 between the United Aircraft Corporation Research Laboratories and NASA pertains to an analytical evaluation of the performance potential of electric propulsion systems using SNAP-8 and SNAP-50 power supplies for primary propulsion of unmanned scientific and logistic space vehicles. During the first two quarters of this study, the application of the SNAP-50 power system to the solar and Saturn probe missions, the Venus orbiter mission, and the lunar logistic supply operation was investigated. In all of these mission studies, use of the Saturn IB launcher was assumed.

During the third quarter, the application of SNAP-50 to the Jupiter probe mission was considered, again assuming the use of the Saturn IB launcher. Also, the study of the lunar logistic supply operation was extended to include the use of the Saturn V as the launcher. For this mission, a performance comparison of electric propulsion with the solid-core nuclear rocket was made, and the results are presented in terms of launch rate and weight launched into earth orbit per unit weight delivered to the lunar surface as functions of the lunar supply rate. Author

### CONCLUSIONS

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The performance comparisons which form the basis of these conclusions are based on the use of the SNAP-50 with a mercury-bombardment ion thruster and an advanced metallic-core nuclear rocket for the Jupiter probe mission. For the lunar logistic supply operation, the results are based on the use of multiple SNAP-50's coupled in parallel and a graphite-core nuclear rocket.

1. For mission times greater than 620 days, greater payloads can be achieved for the Jupiter probe mission by using an electric propulsion system with a powerplant of 30 lb/kwe specific weight than by using a

nuclear rocket of 800 sec specific impulse. At 10 lb/kwe, electric propulsion is superior for mission times greater than about 300 days.

2. Electric propulsion requires fewer Saturn V launches per month to maintain a given supply rate in the lunar logistic supply operation than are required by the nuclear rocket.

3. In supplying two payload deliveries over a powerplant lifetime of 10,000 hr, the electric propulsion system provides greater performance, in terms of earth orbital weight per unit payload weight, than a nuclear rocket of 800 sec specific impulse even up to a specific powerplant weight of approximately 40 lb/kwe. For three payload deliveries in 10,000 hr, electric propulsion is superior up to a specific weight of about 25 lb/kwe.

## INTRODUCTION

The background of this study and the basic assumptions under which it is being carried out were presented in the two previous quarterly progress reports (Refs. 1 and 2). The same general assumptions and ground rules have been maintained in the study during the past quarter except for modifications to the specific ground rules of the lunar logistic supply operation. These changes and their effect on the results are discussed in the presentation of the results.

## JUPITER PROBE MISSION

### Definition of Mission and Assumptions

#### Electric Propulsion

It is assumed that a 28,000 lb electrically propelled space vehicle is launched into a 300 n mi circular earth satellite orbit by a Saturn IB launcher. The space vehicle follows a spiral escape trajectory in the strong gravitational field of the earth and then a heliocentric transfer trajectory consisting of an initial powered phase followed by a coast to Jupiter. The orbits of earth and Jupiter are assumed circular and coplanar. It is assumed that the vehicle operates at constant specific impulse (i.e., constant thrust) but that the thrust may be turned off altogether. The constant-thrust acceleration plus optimum-coast heliocentric trajectory data were obtained from Ref. 3.

Nuclear Rocket Propulsion

The mission and the initial vehicle gross weight for nuclear rocket propulsion are exactly the same as those for electric propulsion. The transfer consists of an escape hyperbola at earth, with a perigee of 300 n mi altitude, followed by a heliocentric elliptic or hyperbolic transfer.

A so-called $1\frac{1}{2}$-stage advanced tungsten-core nuclear rocket vehicle is employed for this comparison of electric and nuclear rocket propulsion. This vehicle, which uses liquid hydrogen as the propellant, is the same as that discussed in Ref. 4. Not under development at present, the advanced nuclear rocket engine is assumed to have a weight of 1000 lb plus 1 lb for every 45 lb of thrust at a specific impulse of 800 sec. In the powered take-off from the initial earth orbit, a propellant tank is jettisoned after half the required velocity change is achieved. The same engine provides the thrust to achieve the remaining velocity change using propellant contained in a second tank.

## Results

For electric-propulsion systems, payload fraction as a function of mission time for given values of power conversion efficiency ($\eta$) and specific impulse ($I$) is presented in Figs. 1 through 9. These generally applicable plots are presented for assumed powerplant specific weights of 10, 20, and 30 lb/kwe and powerplant fractions of 0.20, 0.25, and 0.30. For any specific thruster there would be a characteristic variation of power conversion efficiency with specific impulse. Typical variations are presented in Fig. 10 for a mercury-bombardment ion engine and an arc-jet. Any such variation can be plotted on the generalized grids of Figs. 1 through 9 to obtain payload fraction as a function of mission time for that particular thruster.

Also for a given thruster and powerplant specific weight, optimum values of powerplant fraction and specific impulse could be determined by superimposing the plots of payload versus mission time for the three powerplant fractions. The envelope of these curves would give the maximum possible payload at any given mission time. The optimum value of powerplant fraction has been found to be in the vicinity of 0.25 and is nearly invariant with specific powerplant weight and mission.

Superimposed on Figs. 1 through 9 are payload fraction curves for the $1\frac{1}{2}$-stage solid-core nuclear rocket for general comparison with the electric propulsion performance. The initial thrust-to-weight ratio is 0.5 with assumed values of specific impulse of 700, 800, and 900 sec. To obtain a specific comparison of electric propulsion and nuclear rocket performance, the grids of Figs. 2, 5, and 8 and the efficiency variation of Fig. 10 for the mercury-bombardment ion engine were employed to produce the electric-propulsion payload-fraction curves of Fig. 11. This figure shows that either the SNAP-50 powerplant lifetime must be greater than 10,000 hr (417 days) or its specific weight must be below the 15 to 20 lb/kwe range in order for electric propulsion to be superior to solid-core nuclear rocket propulsion for this mission.

## LUNAR LOGISTIC SUPPLY OPERATION

### Analysis

Since this operation has been previously described in detail in Ref. 2, a detailed description is not repeated here. In Ref. 2 specific cost results (cost per unit weight of payload) were presented for a lunar supply operation based on the Saturn I and Saturn IB launchers. An operation using the Saturn V as the launcher has been studied this quarter, using essentially the same method of analysis as that presented in Appendix I of Ref. 2. Two of the basic ground rules, however, have been changed. After the useful lifetime of the electric propulsion unit has been reached, it is left in lunar orbit rather than being returned to earth orbit to save the fuel required for the return trip. A greater payload may be carried on the last one-way trip than that which is carried on the round-trips.

A second change in ground rules assumes that the payload would be placed on the lunar surface instead of only in lunar orbit. For the purpose of deorbiting the payloads to the surface, a chemical rocket using storable $N_2O_4$ and aerozine with a specific impulse of 312 sec is employed both for the electric propulsion and the nuclear rocket systems. Storable propellant is used instead of cryogenic hydrogen and oxygen to avoid boil-off losses. The nuclear rocket vehicle could land directly on the lunar surface but in so doing could present a possibly unacceptable radiation hazard. These changes in ground rules require some modification of the analysis but not enough to warrant its presentation in this progress report.

#### Electric Propulsion

The electrically propelled vehicles are again considered to be composed of two modules: one containing the propulsion system and a structure frame, and the other containing the propellant, tankage, and payload. In the analysis the initial gross weight of each vehicle is considered to be the same for every trip. Each module of the electrically propelled vehicle is launched separately by one or more Saturn V's. Both an arc-jet and a mercury-bombardment ion thruster are considered for the operation. The assumed variations of power conversion efficiency with specific impulse for the two thrusters are shown in Fig. 10. The arc-jet curve is taken from Ref. 5, and the ion-engine curve is taken from Ref. 6.

The results for the lunar mission are presented in Figs. 12 through 18, which show specific supply rate (i.e., payload weight per day divided by initial vehicle gross weight) as a function of powerplant fraction and the number of payload deliveries per vehicle. A set of these curves is presented for the mercury-bombardment ion thruster for both 10,000 hr and 15,000 hr powerplant lifetime at powerplant specific weights of 10, 20, and 30 lb/kwe. Because the arc-jet has proven itself inferior to the ion thruster for this mission, only a single set of curves, for a powerplant lifetime of 10,000 hr and a powerplant specific weight of 20 lb/kwe, are presented.

Any point on the curves of Figs. 12 through 18 represents an operating point having compatible values of powerplant lifetime, trip time, number of payload deliveries, specific impulse, and powerplant fraction. The only restriction on the use of these plots is that the allowable powerplant fractions must be such that each of the two modules of the vehicle requires the full launching capacity of an integral number of Saturn V's. The powerplant fraction to be used in Figs. 12 through 18 is obtained by subtracting the structure plus thruster fraction (assumed to be 0.04) from the fraction of total vehicle represented by the first module. For example, one Saturn V launching the first module and a second Saturn V launching the second module results in a case for which the powerplant fraction is 0.460. The use of one Saturn V for the first module and two for the second results in a powerplant fraction of 0.293, etc.

The first combination that was investigated was a single Saturn V for each of the modules. As can be seen in Figs. 12 through 18, the corresponding powerplant fraction of 0.460 is too high, since it results in operation well past the peak of maximum supply rate for any number of payload deliveries per vehicle for both the mercury-bombardment ion engine and the arc-jet. Therefore another choice was made involving one Saturn V to launch the first module and two others to launch the second (powerplant fraction of 0.293). Although this combination possibly increases the difficulty of the rendezvous problem and also requires a larger vehicle, the increase in performance would probably warrant the acceptance of these additional problems.

An alternate scheme which was not studied may have potential. This scheme would involve the initial launch of both modules with a single Saturn V, and to launch each successive second module also with a Saturn V. Since this scheme results in different initial gross weights, the results of the present study cannot be applied.

As can be seen from Fig. 18 a powerplant fraction of 0.293 is still too large for efficient operation of the arc-jet. To make the arc-jet competitive with the mercury-bombardment ion engine, it is necessary to increase the number of Saturn V's to five, with one launching the first module and the other four launching the second module. Although the power requirements of these large vehicles are large (around 10 Mwe at a powerplant specific weight of 20 lb/kwe), it is assumed that this power can be supplied by coupling five SNAP-50 powerplants in parallel. If the packaging problem is too severe, either a 10 Mwe powerplant or a smaller vehicle would have to be used.

One further comment regarding Figs. 12 through 18 may be useful. The supply rates of these figures are based on the use of the storable propellant $N_2O_4$-aerozine for deorbiting. The ratio of payload weight on the lunar surface to weight in lunar orbit for this propellant is approximately 0.533. The corresponding supply rate to the lunar orbit can be determined by multiplying the supply rate to the surface by 1/0.533.

Nuclear Rocket Propulsion

Unlike the powerplant of the electrically propelled vehicle, the nuclear rocket engine weight is a small part of the total vehicle weight. For this mission a NERVA-type graphite-core nuclear rocket is employed which has an assumed weight of 4500 lb plus 1 lb for every 45 lb of rocket thrust at a specific impulse of 800 sec.

Initially the whole vehicle is launched by a Saturn V into the standard 300 n mi circular orbit. For the next lunar trip, the Saturn V would launch the same weight of payload, propellant, and tankage into an orbit of slightly higher energy due to the fact that the engine and structure frame are already in orbit. On the final one-way trip of the vehicle, a larger payload can be carried, since no propellant would be required for the return trip. Thus, during the life of one vehicle, there are three different sizes of payload carried.

It is necessary to determine the best values of round-trip time and vehicle thrust-to-weight ratio for the nuclear vehicles. These computations have been done for a specific impulse of 800 sec, and the results are assumed to hold for other values of specific impulse. A good figure of merit for performance in this mission is the weight launched into earth orbit per unit payload weight delivered to the lunar surface. It is desirable to minimize this figure of merit. This quantity is at a minimum value at a thrust-to-weight ratio of around 0.4 for nearly all values of round-trip time. It is very insensitive to thrust-to-weight ratio around the minimum, so a value of 0.5 was chosen in order that gravity $\Delta V$ losses may be neglected in the calculations without loss of accuracy.

At any thrust-to-weight ratio, the figure of merit is minimum for a round-trip time corresponding to minimum-energy transfers. This time is about 10.5 days. Therefore, results are presented for a round-trip time of 10.5 days and a thrust-to-weight ratio of 0.5. It should be emphasized that the 10.5 day round-trip time minimizes only the performance figure of merit, whereas a shorter round-trip time would minimize the specific cost of the operation. Since the comparison is being made here on the basis of performance, the longer time is employed. Results of a minimum specific cost comparison will be presented in the final report.

## Results

The required Saturn V launch rate (number of launches per month) is shown in Fig. 19 as a function of the supply rate to the lunar surface for both electric and nuclear-rocket propulsion. Presented in the figure are results for the mercury-bombardment ion engine and the arc-jet for a powerplant specific weight of 20 lb/kwe and a powerplant lifetime of 10,000 hr, and for various numbers of payload deliveries per vehicle. The corresponding values of trip time and specific impulse for the electric propulsion systems presented in this and the following figures are shown in Table I.

For comparison, the same results for the nuclear-rocket system are shown in Figs. 19 through 23 for values of specific impulse of 600, 700, 800, and 900 sec. It is seen that the mercury-bombardment ion engine system requires a lower launch rate than the nuclear-rocket system for both two and three payload deliveries per vehicle. The arc-jet, on the other hand, cannot compete even for the lowest number of payload deliveries considered.

Figure 20 is different from Fig. 19 only in that the powerplant lifetime is increased to 15,000 hr. The result is that a lower launch rate is required for more payload deliveries per vehicle.

Figures 21 and 22 show the same results of required Saturn V launch rate versus supply rate for a powerplant specific weight of 10 lb/kwe. Here all of the electric propulsion curves indicate a lower required launch rate than any of the nuclear rocket curves.

Finally, Fig. 23 presents the same results for a powerplant specific weight of 30 lb/kwe and for powerplant lifetimes of 10,000 hr and 15,000 hr. For 10,000 hr the electric propulsion system is superior for two payload deliveries, while at 15,000 hr it is superior for both two and three payload deliveries.

In Figs. 24 and 25 are plotted powerplant specific weight and nuclear rocket specific impulse against the ratio of weight launched into earth orbit per unit weight delivered to the lunar surface. This figure of merit, as mentioned previously, is a good indicator of system performance.

These two figures clearly show the relative merits of electric and nuclear-rocket propulsion for this mission. It is seen that electric propulsion with a mercury-bombardment ion thruster is superior if a powerplant specific weight of 30 lb/kwe or less can be achieved and if the long round-trip times of the order of 100 to 200 days can be tolerated.

## PROJECTED WORK

During the last three-month phase of the contract, specific cost results for the lunar supply operation using Saturn V launchers, which had not been completed at the time of writing of the present progress report, will be reported.

A study will be launched to determine the applicability of the presently-envisioned SNAP-8 powerplant to the following missions: a 24-hr satellite, a lunar mapping satellite, an out-of-the-ecliptic probe, and a solar probe. An idea for improved electric propulsor efficiency, involving direct thermal heating in the ionizer of cesium contact ion thrusters, will be investigated in connection with the SNAP-8 system.

A draft of the final technical report will be written during the next quarter.

## REPORTING NEW TECHNOLOGY

There are no developments to report under the requirements of the Reporting of New Technology clause in the contract.

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3. Melbourne, W. G.: Unpublished results.

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5. Teem, J. M. and Szego, G. C.: Electric Propulsion and Power. Astronautics and Aerospace Engineering, November 1963.

6. Dennington, R. J., W. J. LeGrey, and R. D. Shattuck: Electric Propulsion for Manned Missions. Presented at the AIAA Meeting on Engineering Problems of Manned Interplanetary Exploration, Palo Alto, California, September 30 to October 1, 1963.

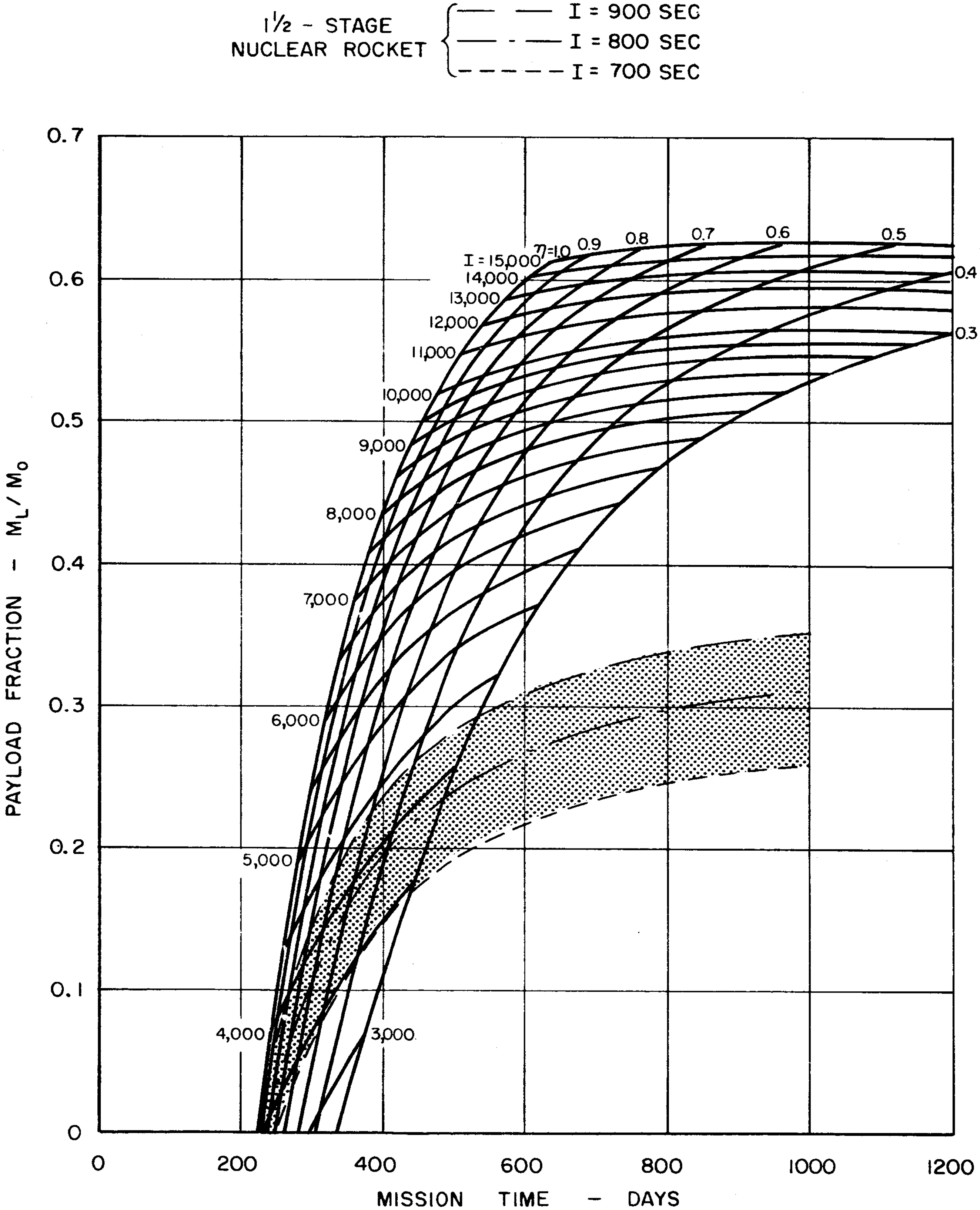
TABLE I

ELECTRIC PROPULSION TRIP TIMES AND SPECIFIC IMPULSE
FOR THE LUNAR LOGISTIC SUPPLY OPERATION

| Thrustor | Powerplant Specific Wt. (lb/KWe) | Powerplant Lifetime (hr) | Number of Payload Deliveries Per Vehicle | Specific Impulse (sec) | Round-Trip Time (days) | Outbound Time (days) |
|---|---|---|---|---|---|---|
| Arcjet | 20 | 10,000 | 2 | 1,750 | 249 | 168 |
| Mercury Bombardment Ion Engine | 10 | 10,000 | 2 | 22,700 | 240 | 177 |
| | | | 3 | 12,900 | 154 | 109 |
| | | | 4 | 8,000 | 114 | 75 |
| | | | 5 | 5,400 | 90 | 57 |
| | 10 | 15,000 | 2 | > 25,000 | 356 | 269 |
| | | | 3 | 21,800 | 229 | 167 |
| | | | 4 | 14,700 | 170 | 115 |
| | | | 5 | 10,400 | 132 | 97 |
| | | | 6 | 7,700 | 110 | 75 |
| | 20 | 10,000 | 2 | 9,400 | 240 | 176 |
| | | | 3 | 4,600 | 154 | 109 |
| | | | 4 | 2,700 | 114 | 75 |
| | 20 | 15,000 | 2 | 16,700 | 356 | 269 |
| | | | 3 | 8,800 | 229 | 167 |
| | | | 4 | 5,500 | 170 | 115 |
| | | | 5 | 3,500 | 132 | 97 |
| | 30 | 10,000 | 2 | 5,300 | 246 | 171 |
| | 30 | 15,000 | 2 | 9,800 | 365 | 260 |
| | | | 3 | 5,000 | 232 | 161 |

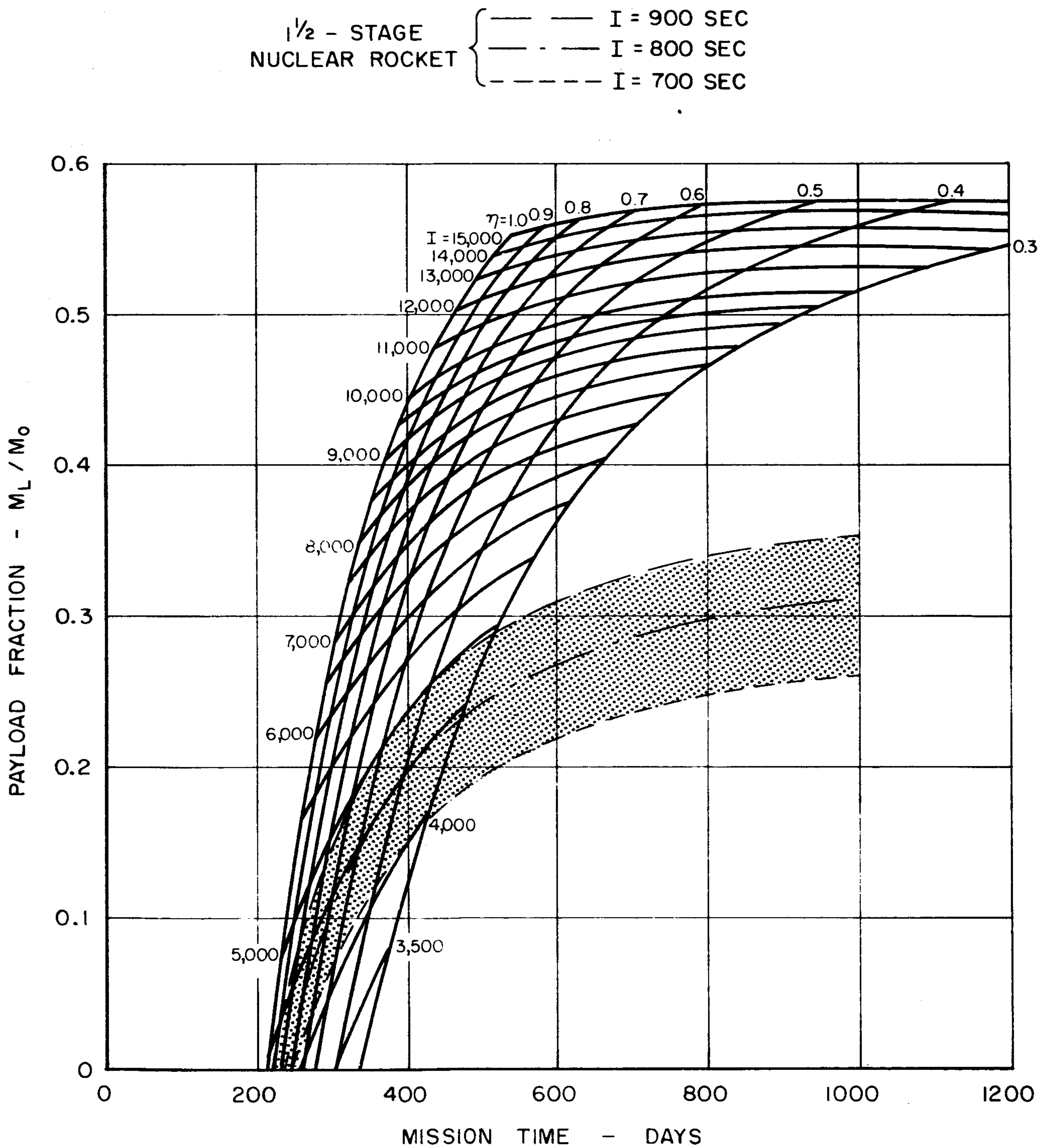
# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
POWERPLANT FRACTION = 0.20
SATURN IB LAUNCHER



# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
POWERPLANT FRACTION = 0.25
SATURN IB LAUNCHER

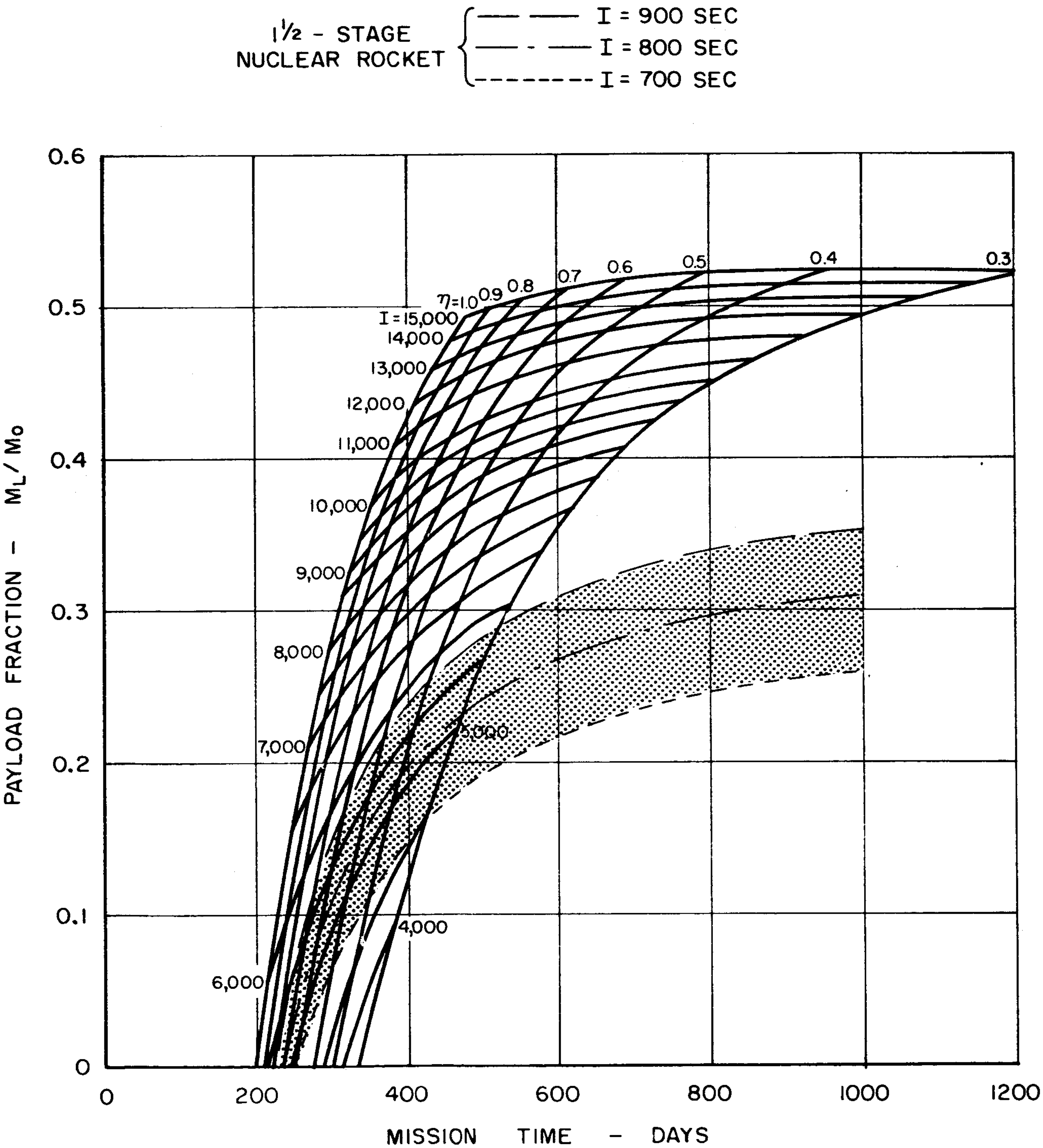


## JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe

POWERPLANT FRACTION = 0.30

SATURN IB LAUNCHER



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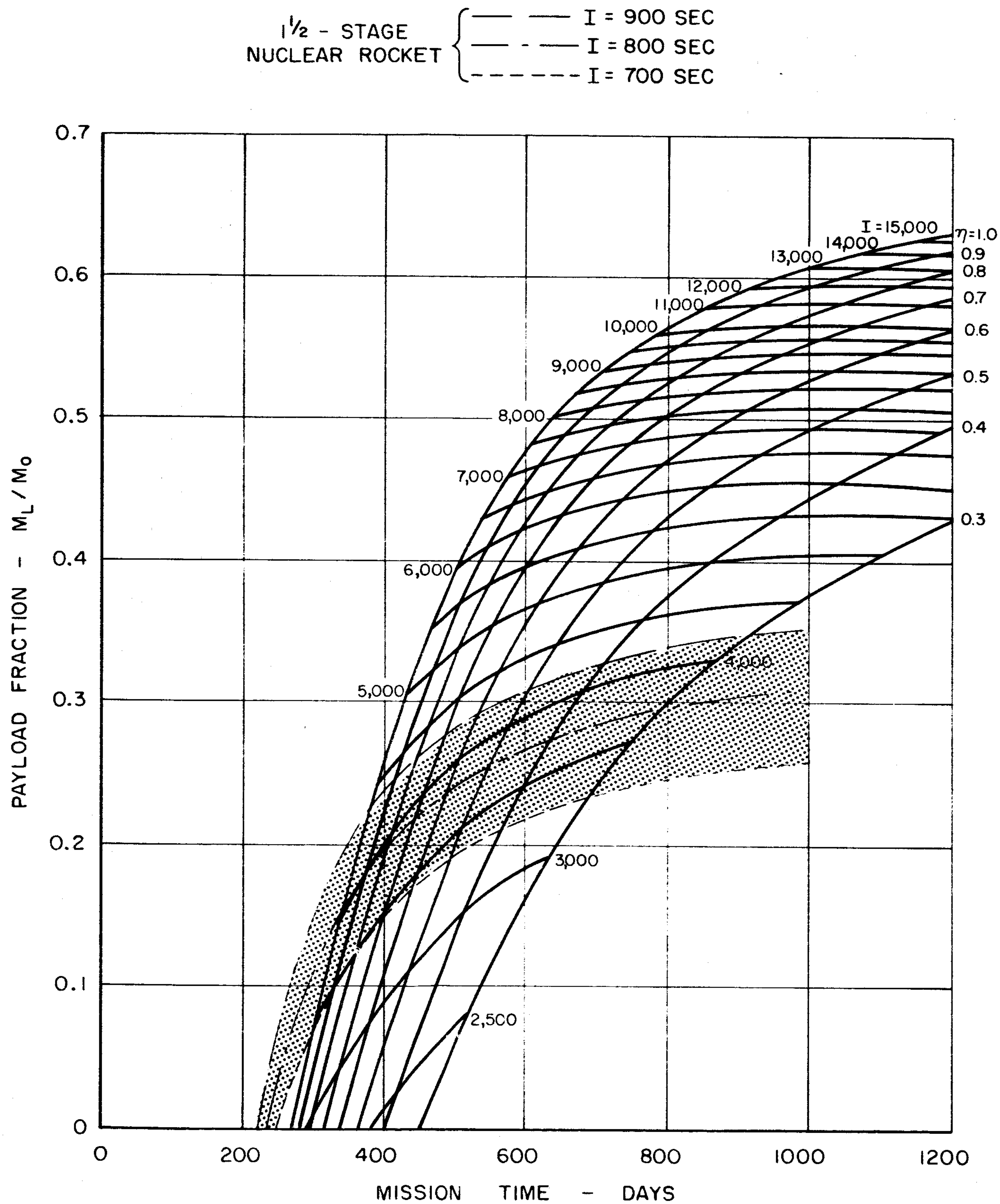
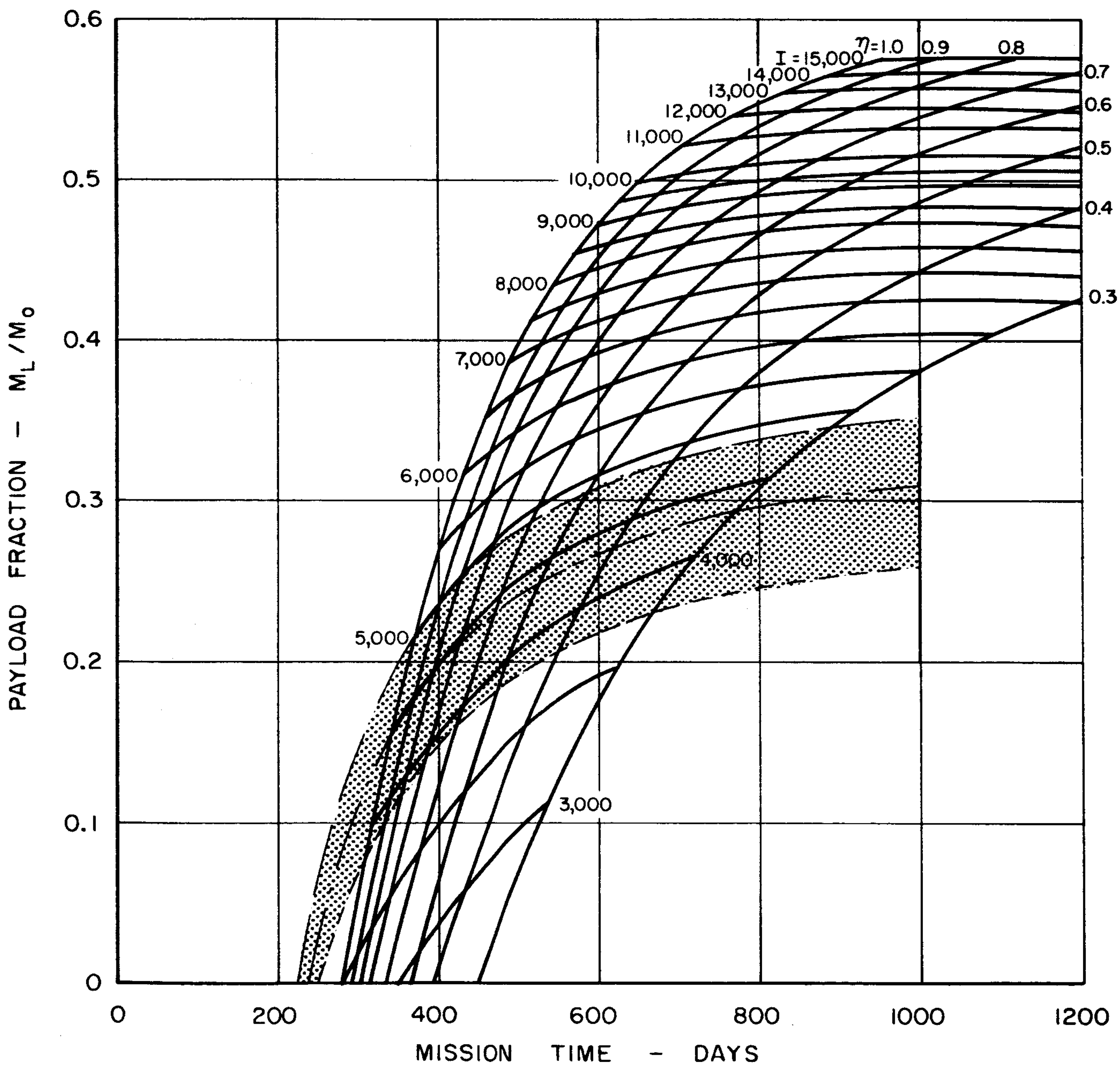
POWERPLANT FRACTION = 0.20
SATURN IB LAUNCHER



# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 20 LB/KWe
POWERPLANT FRACTION = 0.25
SATURN IB LAUNCHER

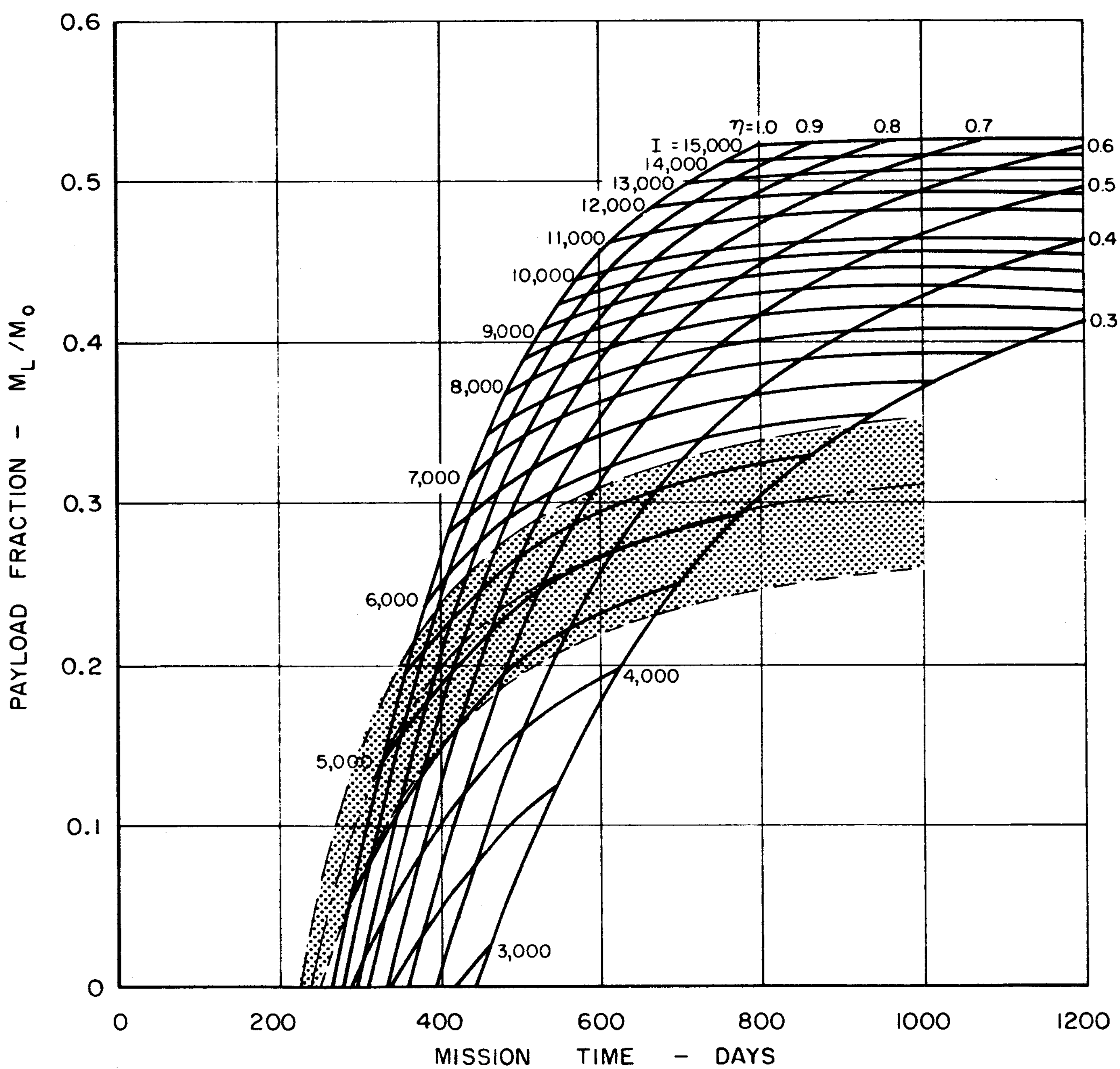
1½ - STAGE NUCLEAR ROCKET
— — I = 900 SEC
— · — I = 800 SEC
- - - - - I = 700 SEC



# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 20 LB/KWe
POWERPLANT FRACTION = 0.30
SATURN IB LAUNCHER

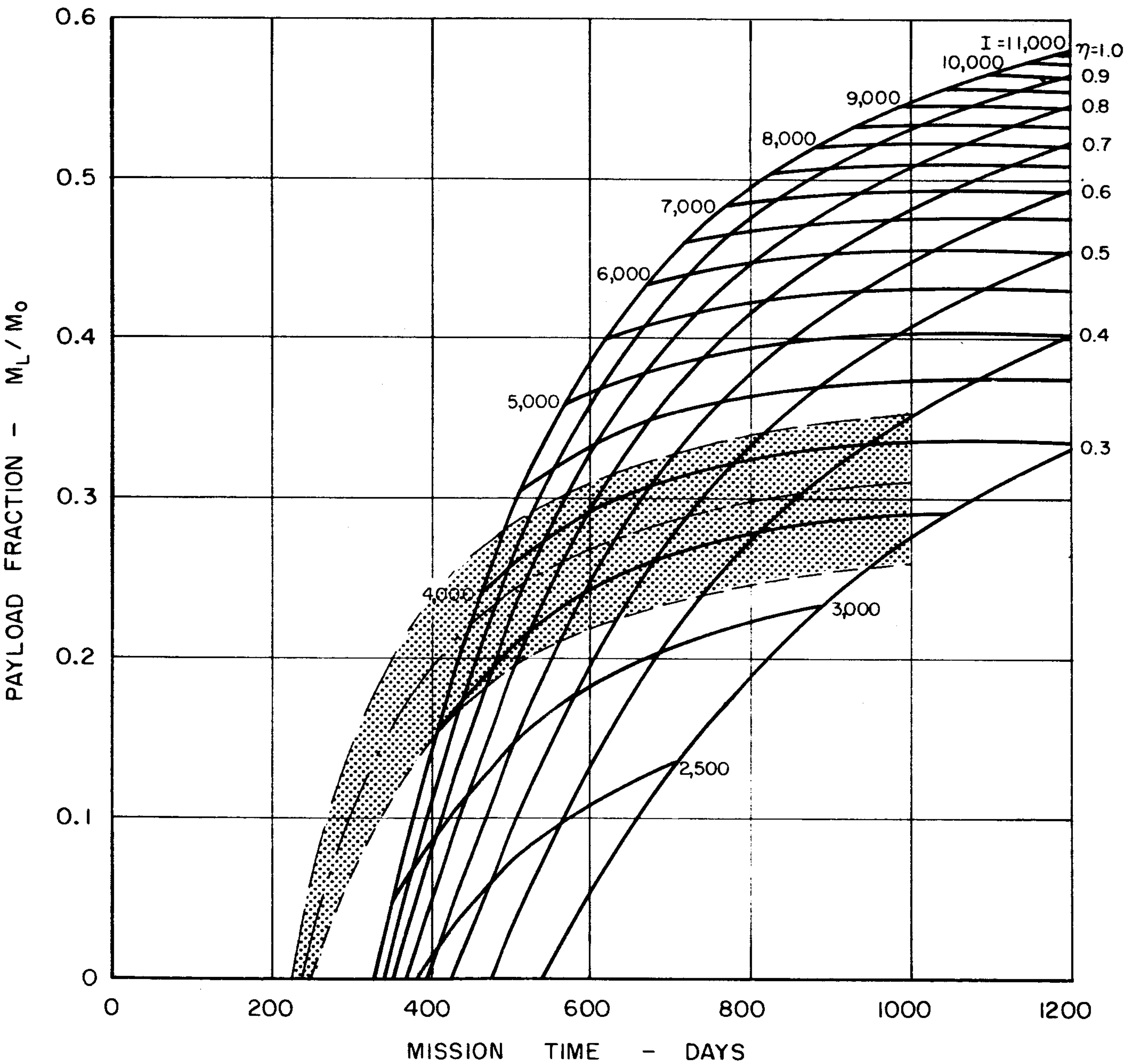
1½ - STAGE NUCLEAR ROCKET
—— —— I = 900 SEC
—— - —— I = 800 SEC
- - - - - - - I = 700 SEC



# JUPITER PROBE MISSION

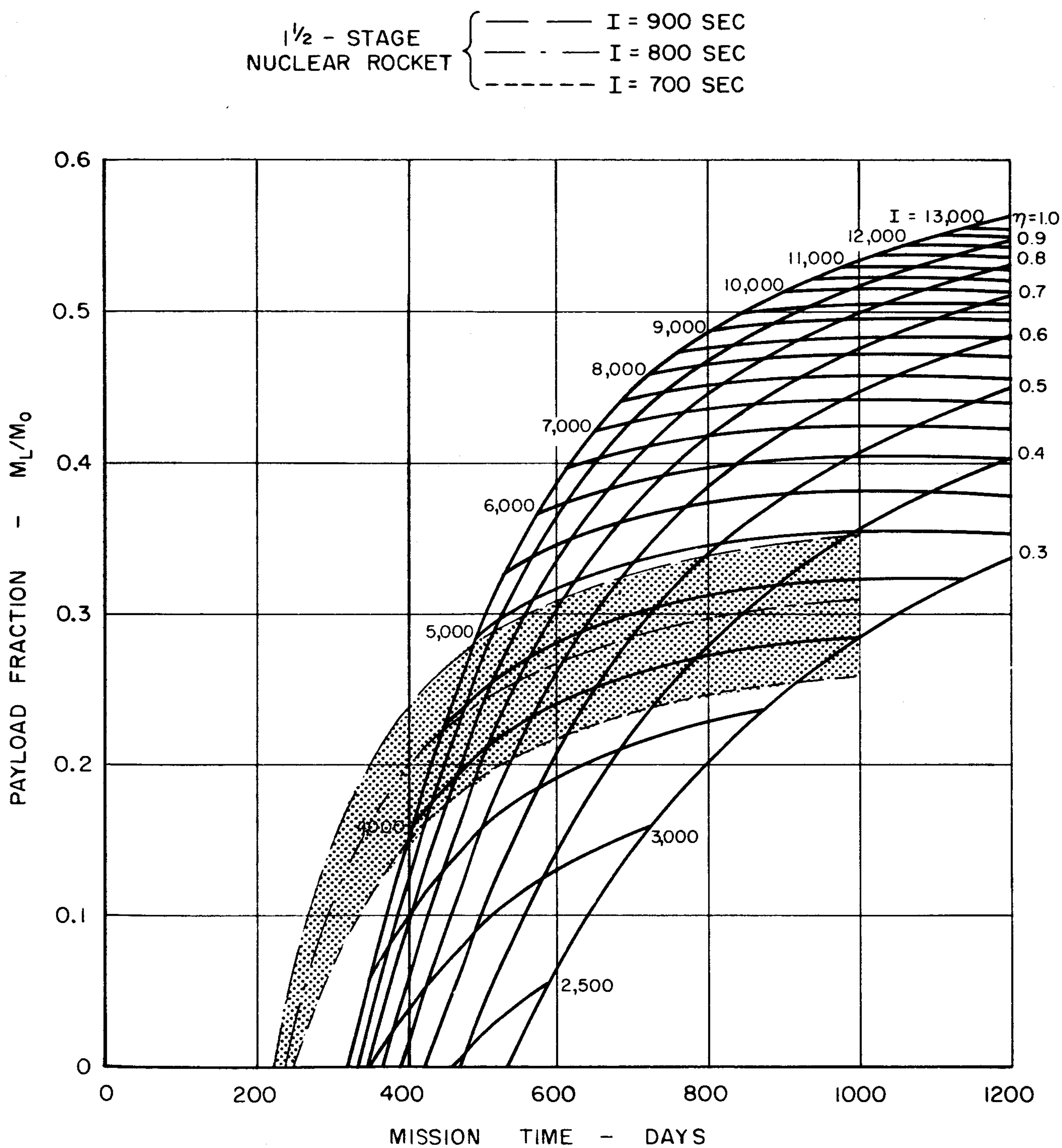
POWERPLANT SPECIFIC WEIGHT = 30 LB/KWe
POWERPLANT FRACTION = 0.20
SATURN IB LAUNCHER

1½ - STAGE NUCLEAR ROCKET
— — I = 900 SEC
— · — I = 800 SEC
- - - - - I = 700 SEC



# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 30 LB/KWe
POWERPLANT FRACTION = 0.25
SATURN IB LAUNCHER



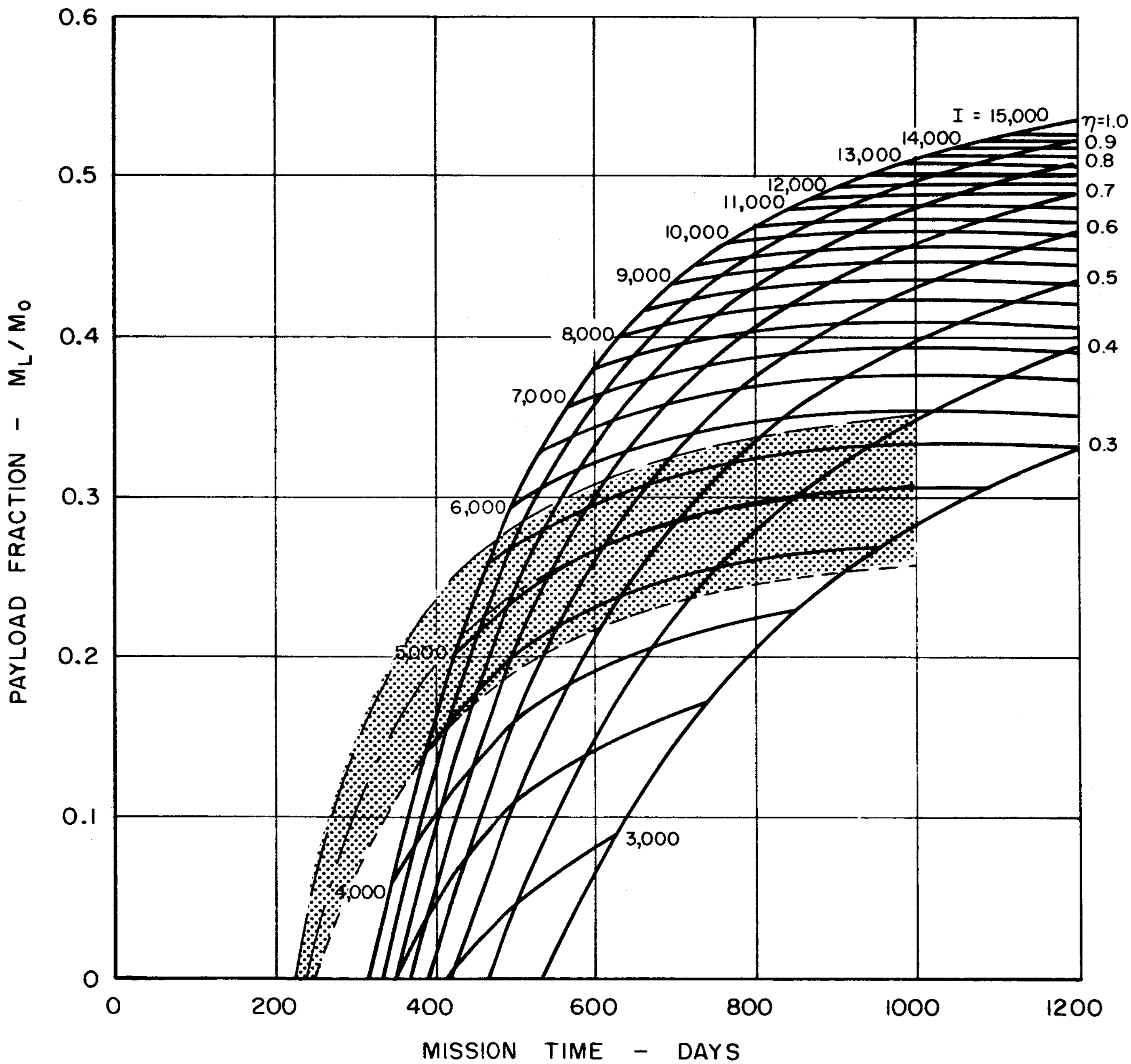
# JUPITER PROBE MISSION

POWERPLANT SPECIFIC WEIGHT = 30 LB/KWe

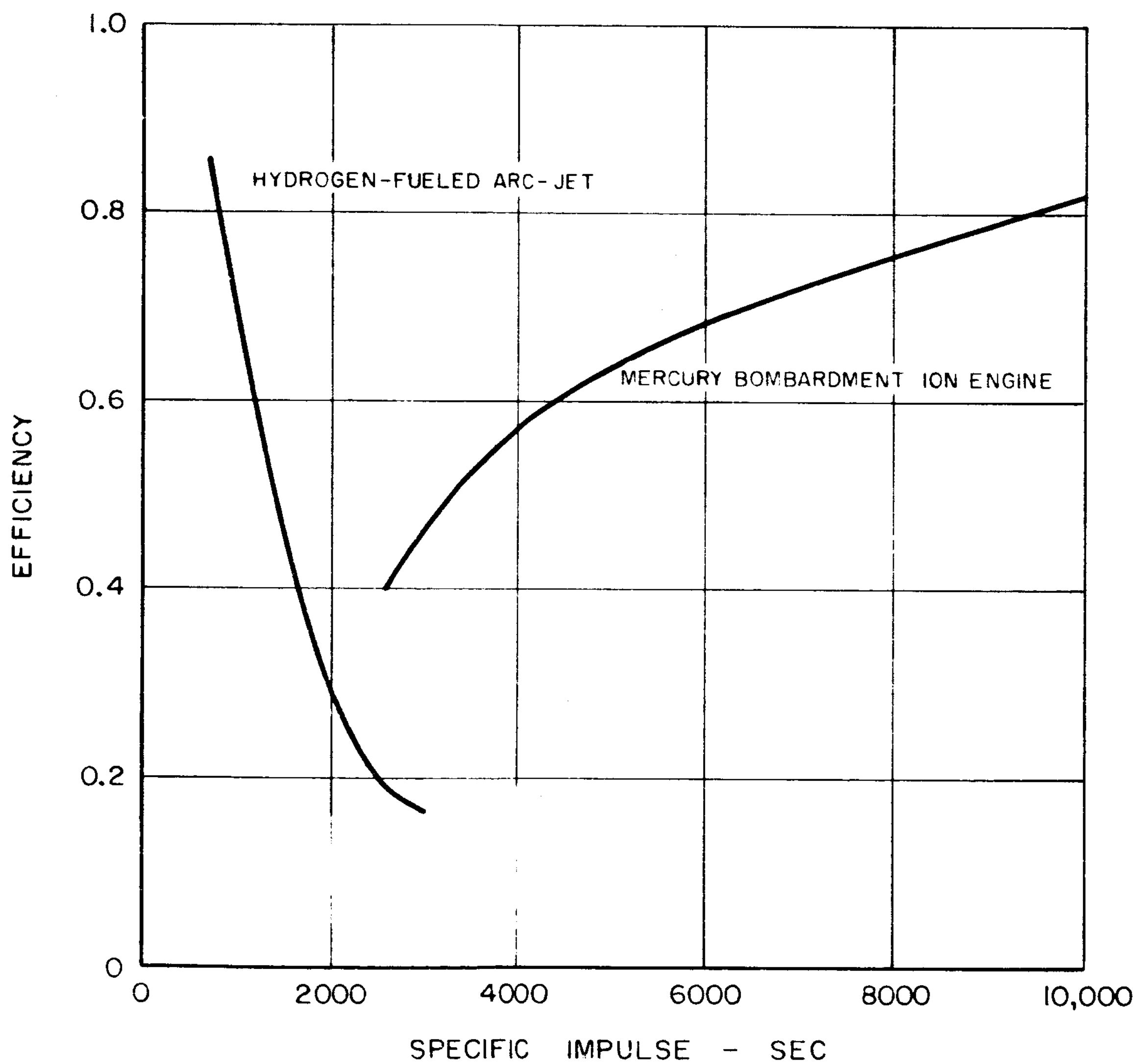
POWERPLANT FRACTION = 0.30

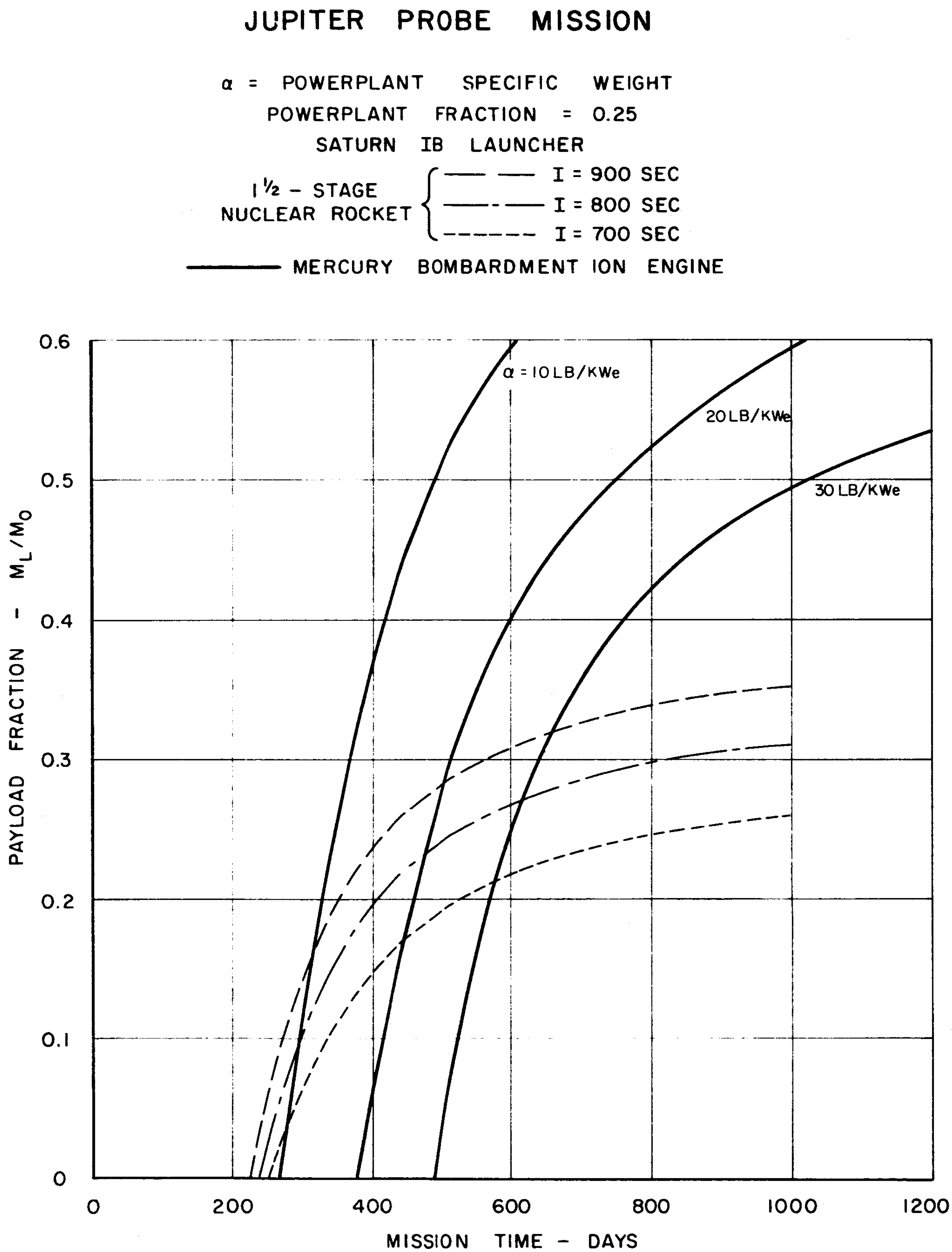
SATURN IB LAUNCHER

1½ - STAGE NUCLEAR ROCKET { —— —— I = 900 SEC; —— · —— I = 800 SEC; - - - - - - - I = 700 SEC }



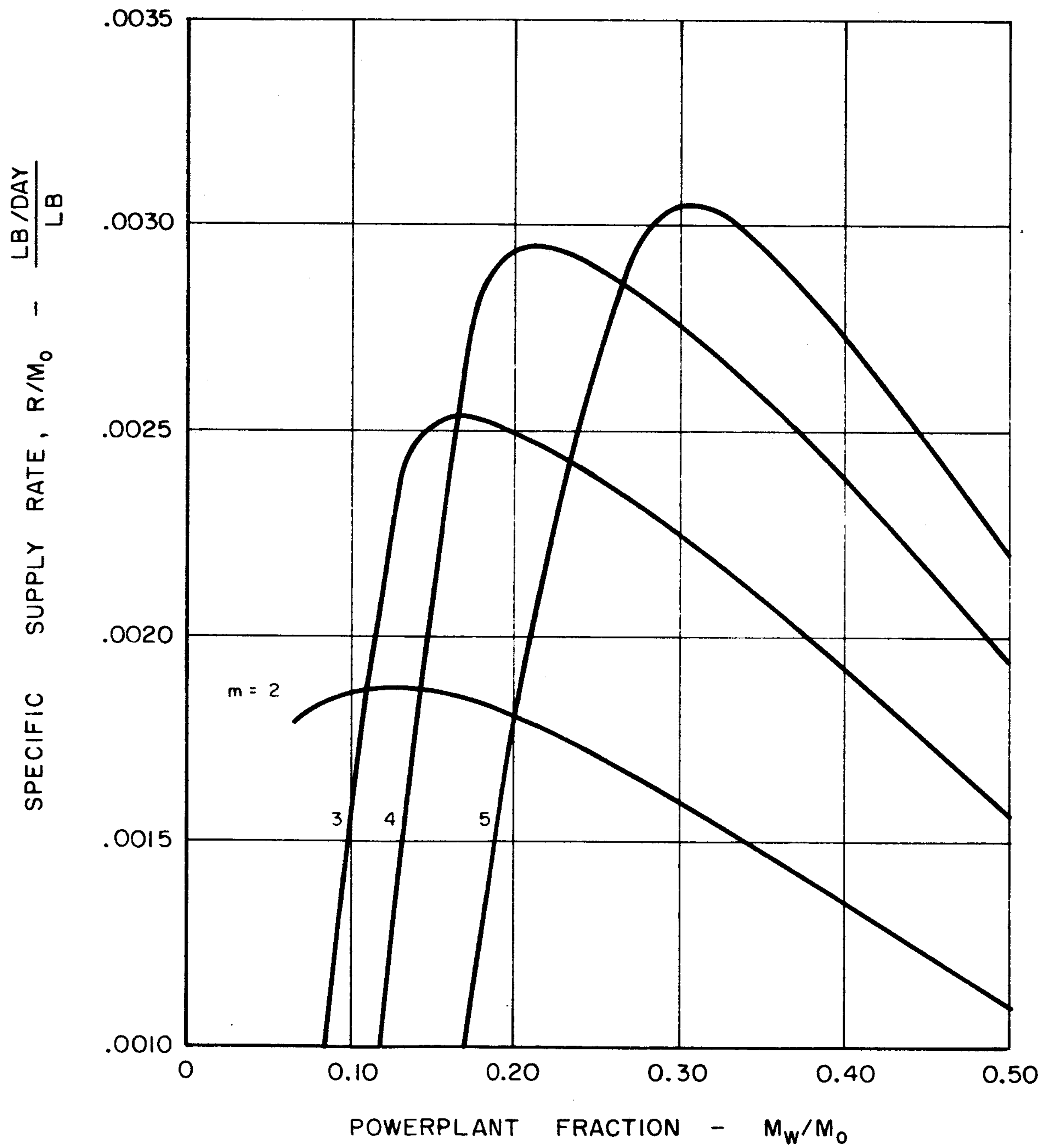
# ESTIMATED VARIATION OF EFFICIENCY WITH SPECIFIC IMPULSE




JUPITER PROBE MISSION
α = POWERPLANT SPECIFIC WEIGHT
POWERPLANT FRACTION = 0.25
SATURN IB LAUNCHER
1 ½ - STAGE NUCLEAR ROCKET
I = 900 SEC
I = 800 SEC
I = 700 SEC
MERCURY BOMBARDMENT ION ENGINE
α = 10 LB/KWe
20 LB/KWe
30 LB/KWe
PAYLOAD FRACTION - $M_L/M_O$
0.6
0.5
0.4
0.3
0.2
0.1
0
0
200
400
600
800
1000
1200
MISSION TIME - DAYS

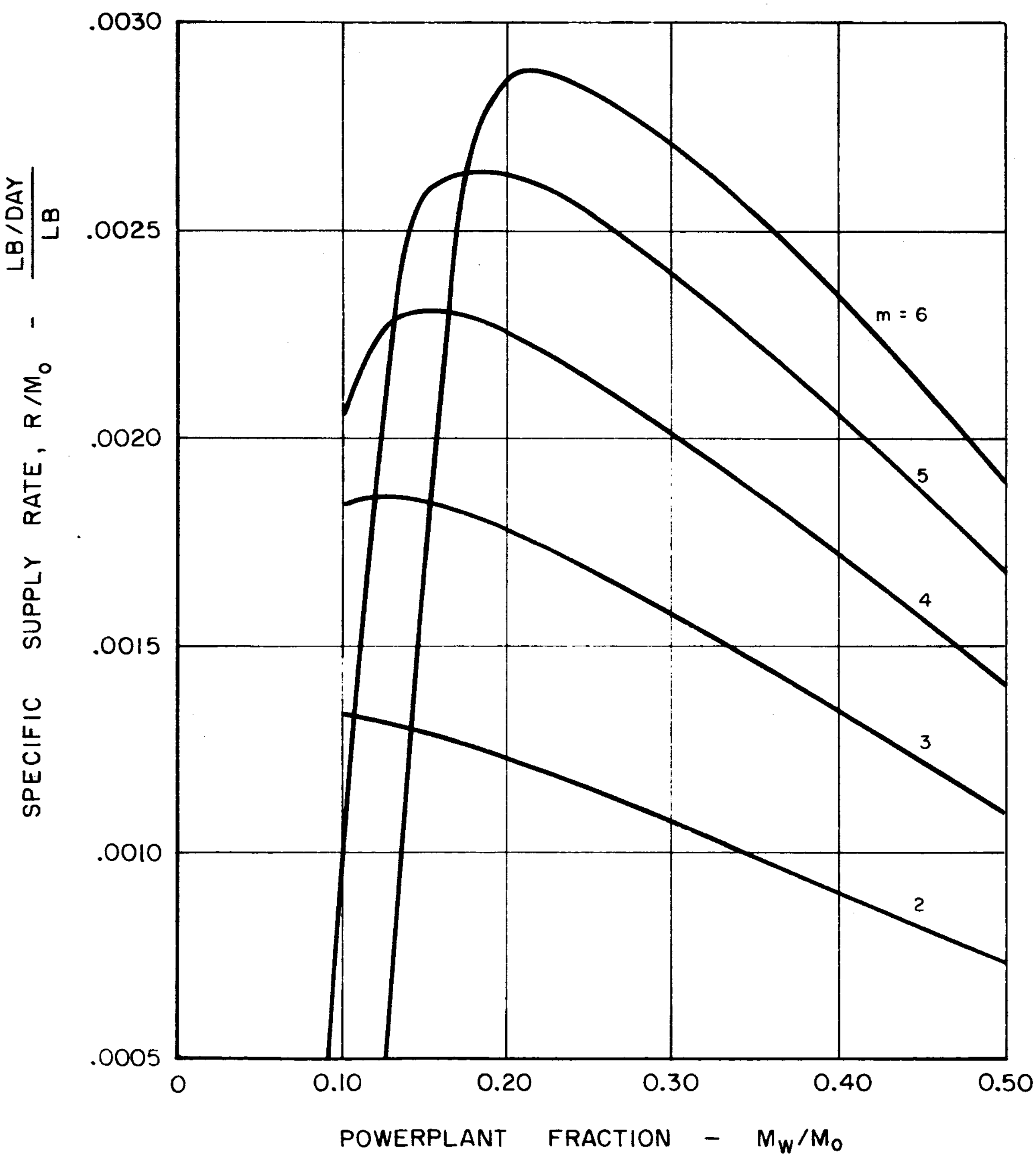
# LUNAR LOGISTIC SUPPLY OPERATION

SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
MERCURY BOMBARDMENT ION ENGINE
POWERPLANT LIFETIME = 10,000 HOURS
POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



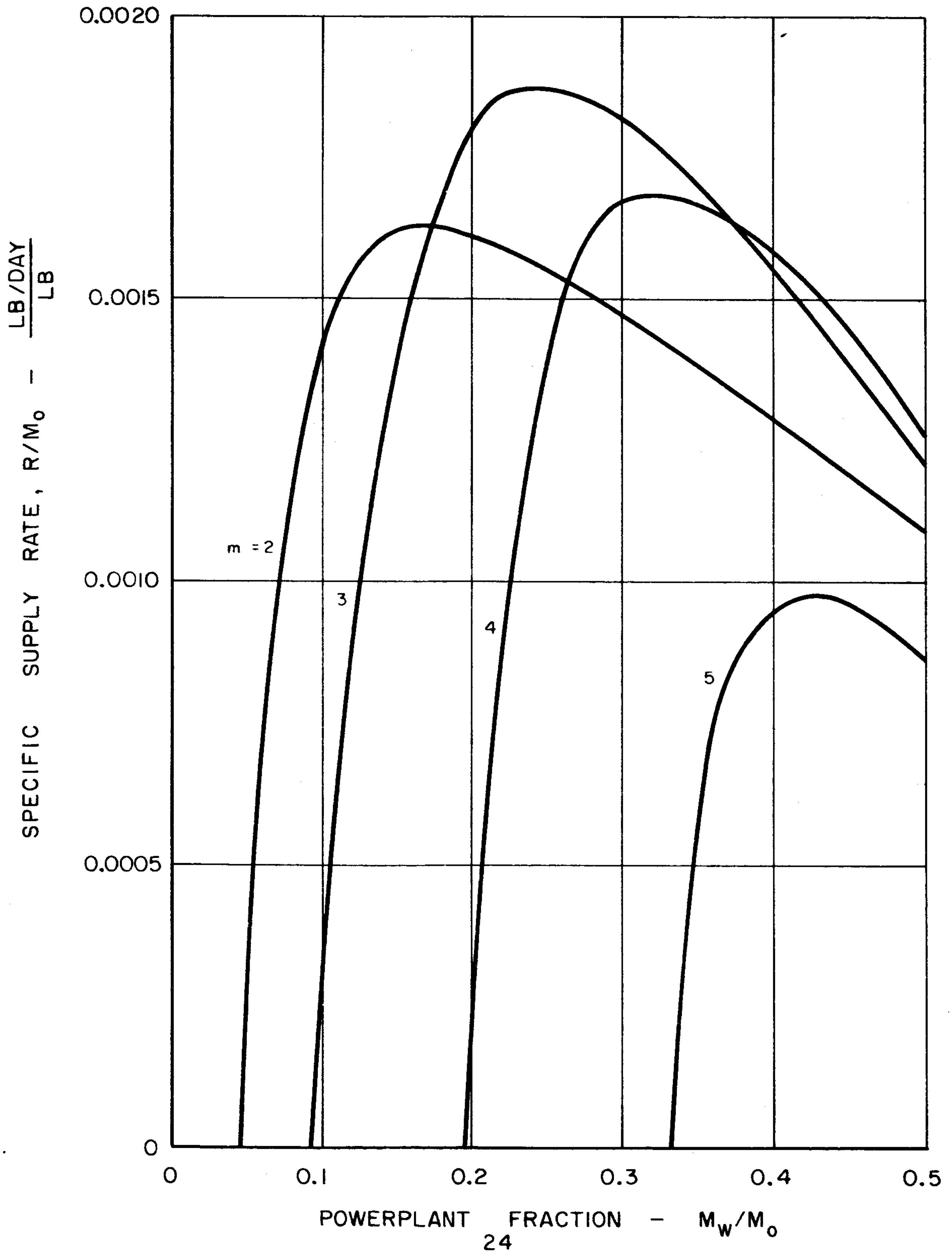
# LUNAR LOGISTIC SUPPLY OPERATION

SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
MERCURY BOMBARDMENT ION ENGINE
POWERPLANT LIFETIME = 15,000 HOURS
POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



# LUNAR LOGISTIC SUPPLY OPERATION

SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
MERCURY BOMBARDMENT ION ENGINE
POWERPLANT LIFETIME = 10,000 HOURS
POWERPLANT SPECIFIC WEIGHT = 20 LB/KWe
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



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POWERPLANT LIFETIME = 15,000 HOURS
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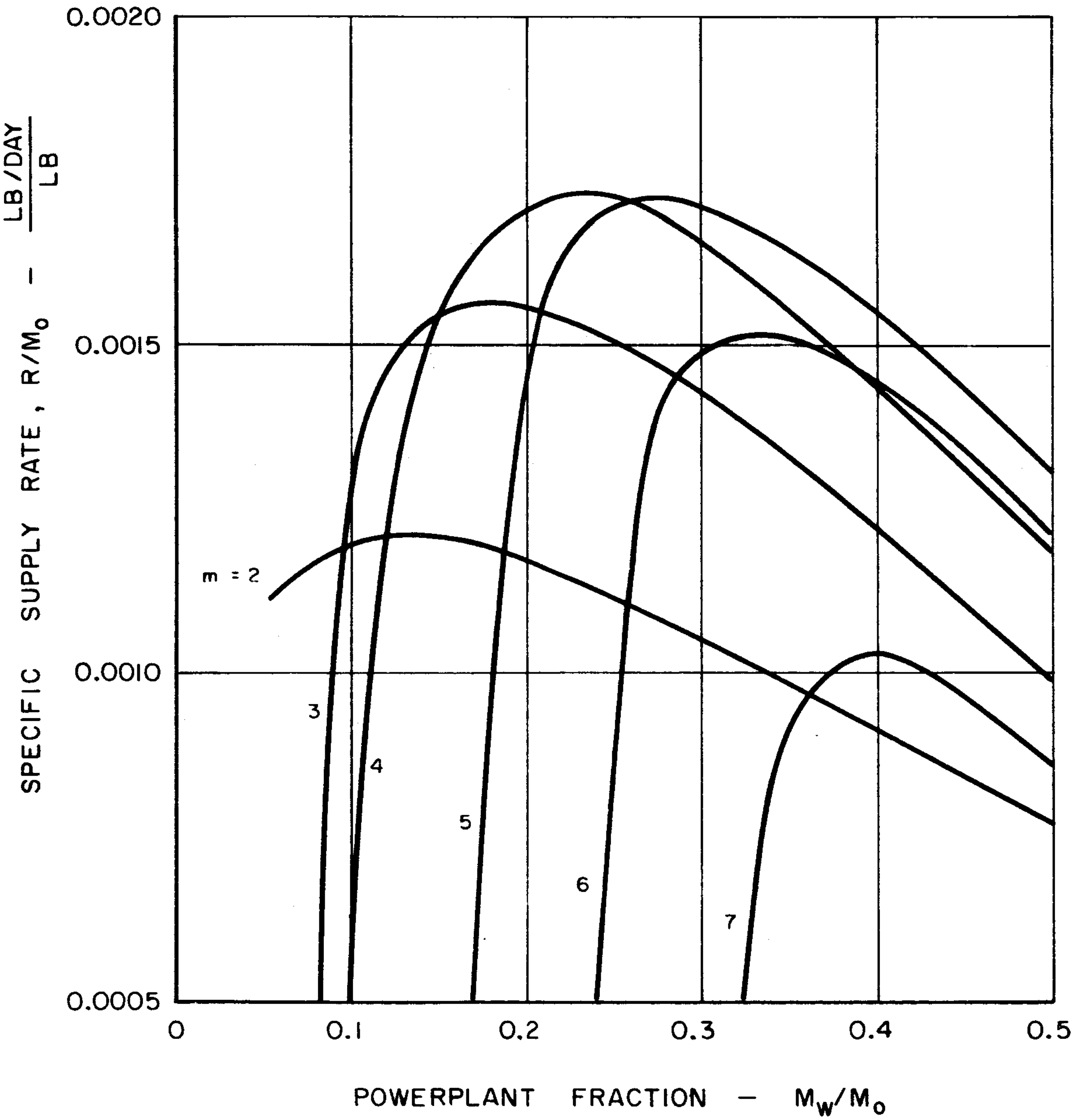
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



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SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
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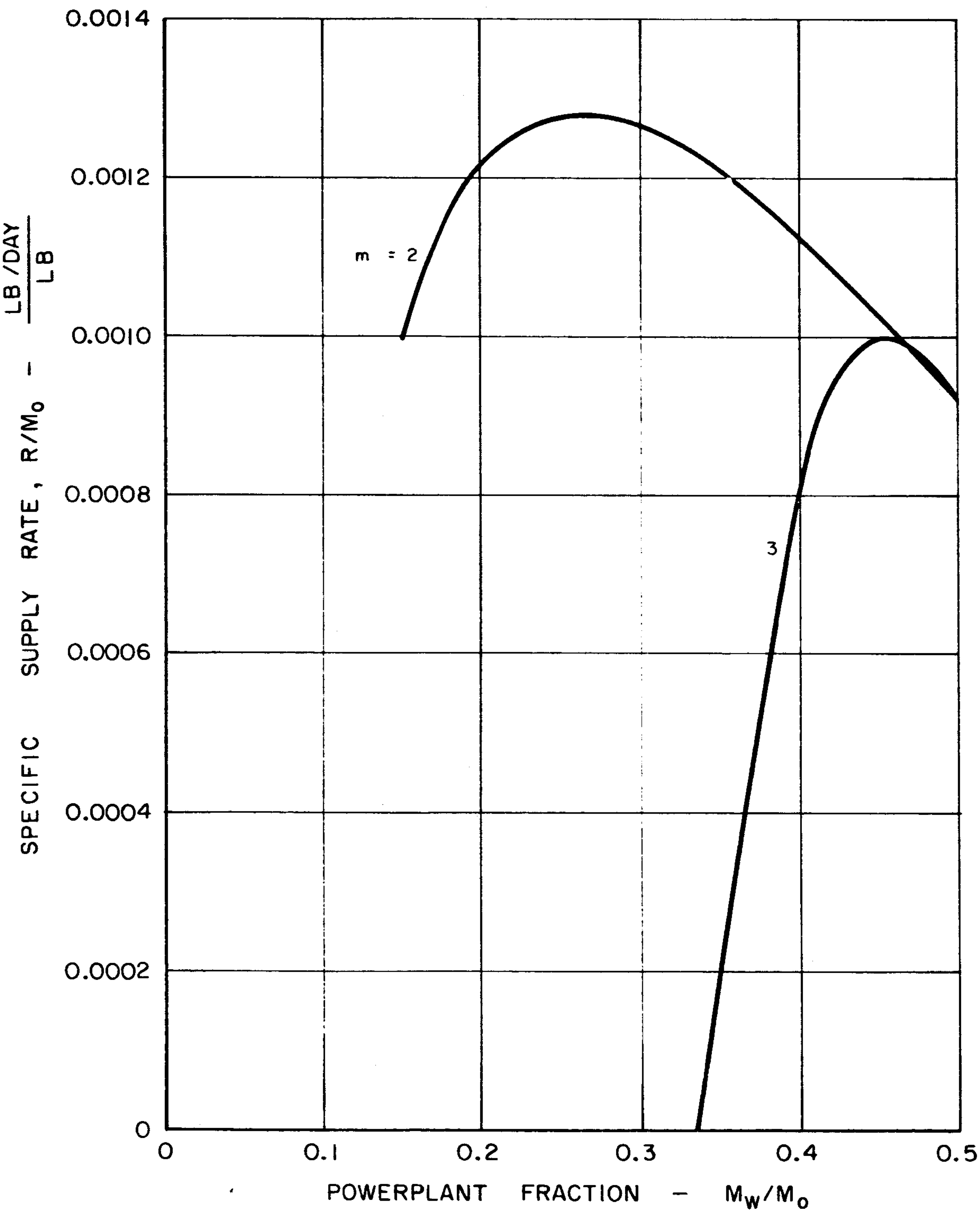

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SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
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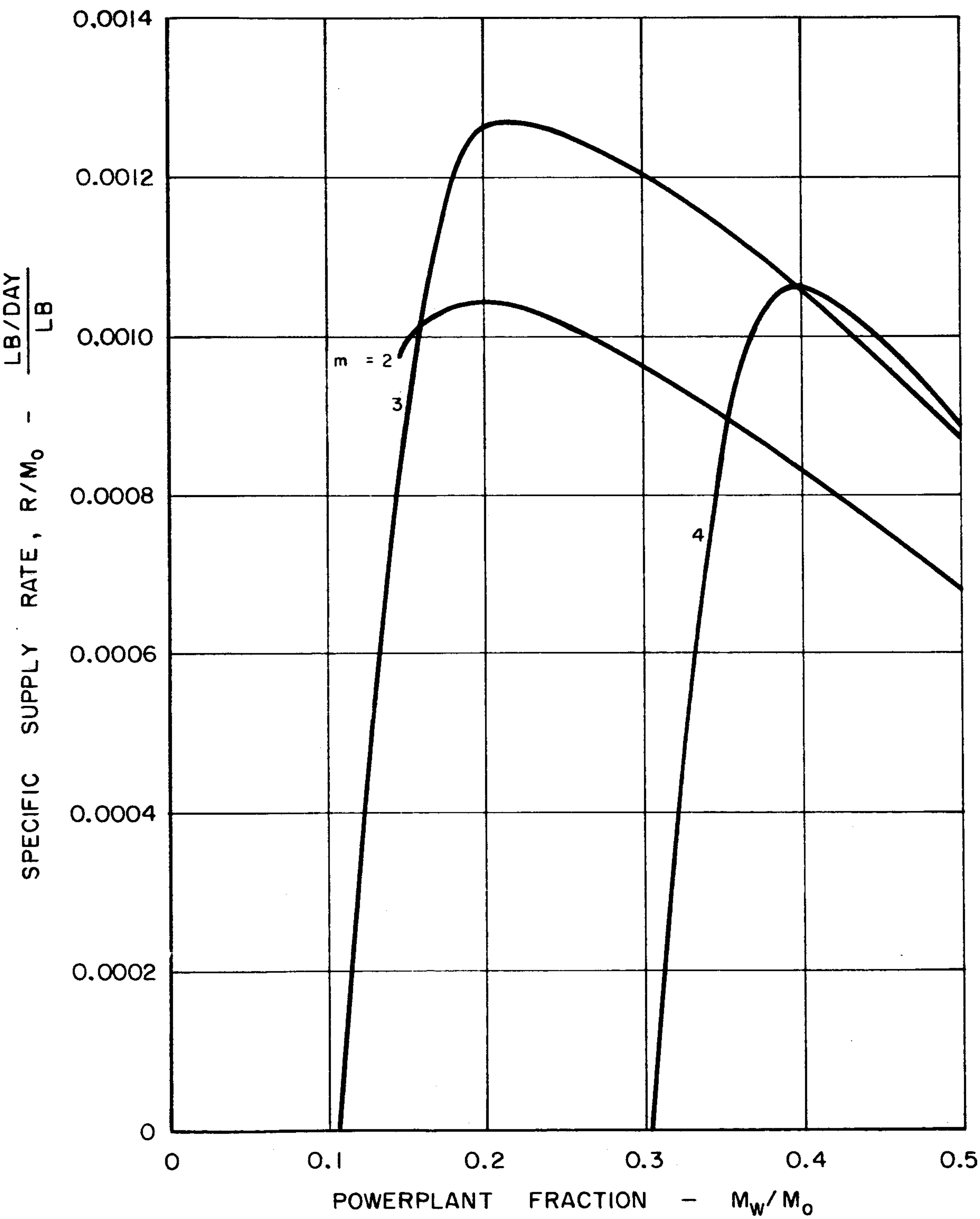
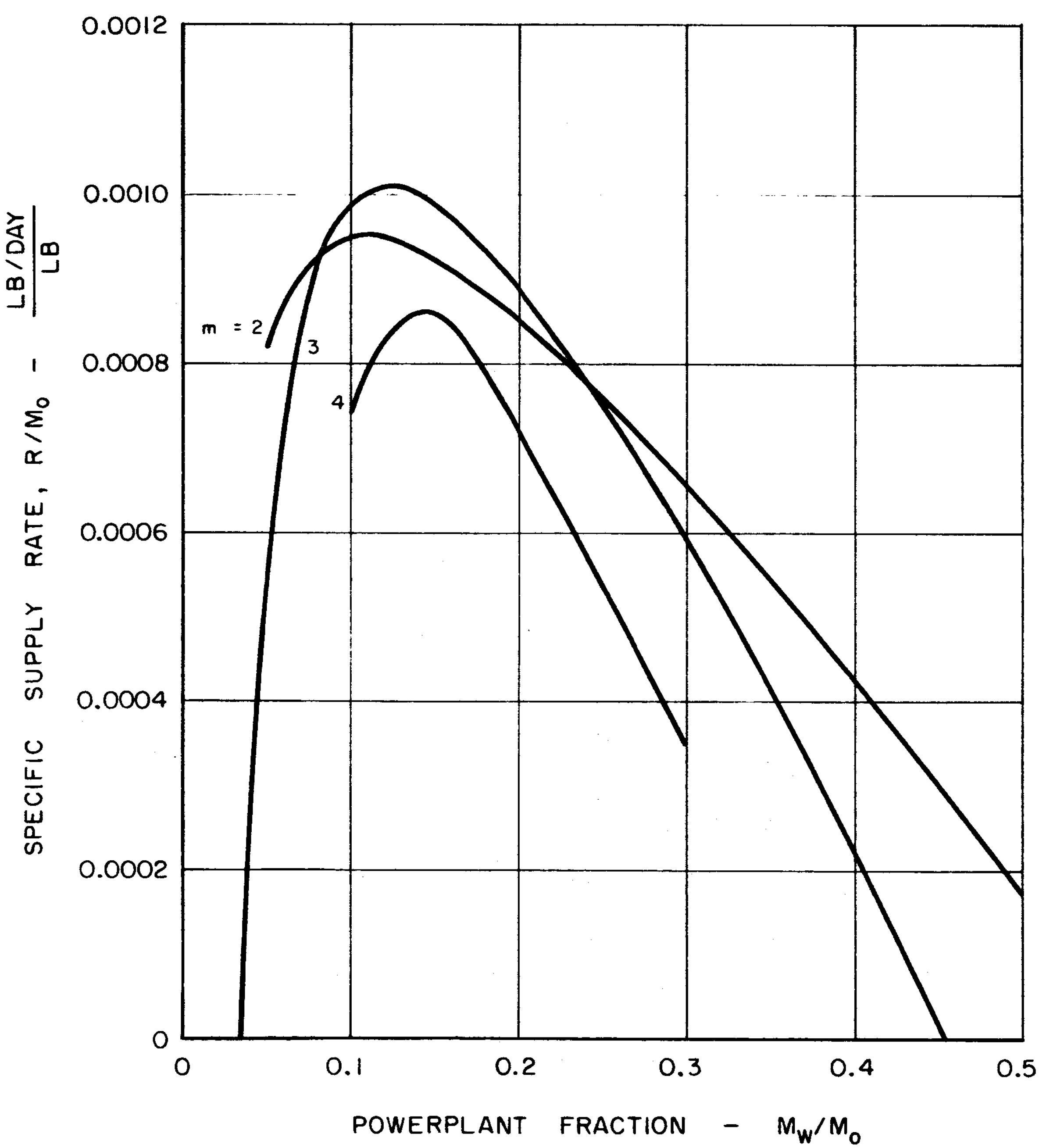
POWERPLANT LIFETIME = 15,000 HOURS
POWERPLANT SPECIFIC WEIGHT = 30 LB/KWe
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



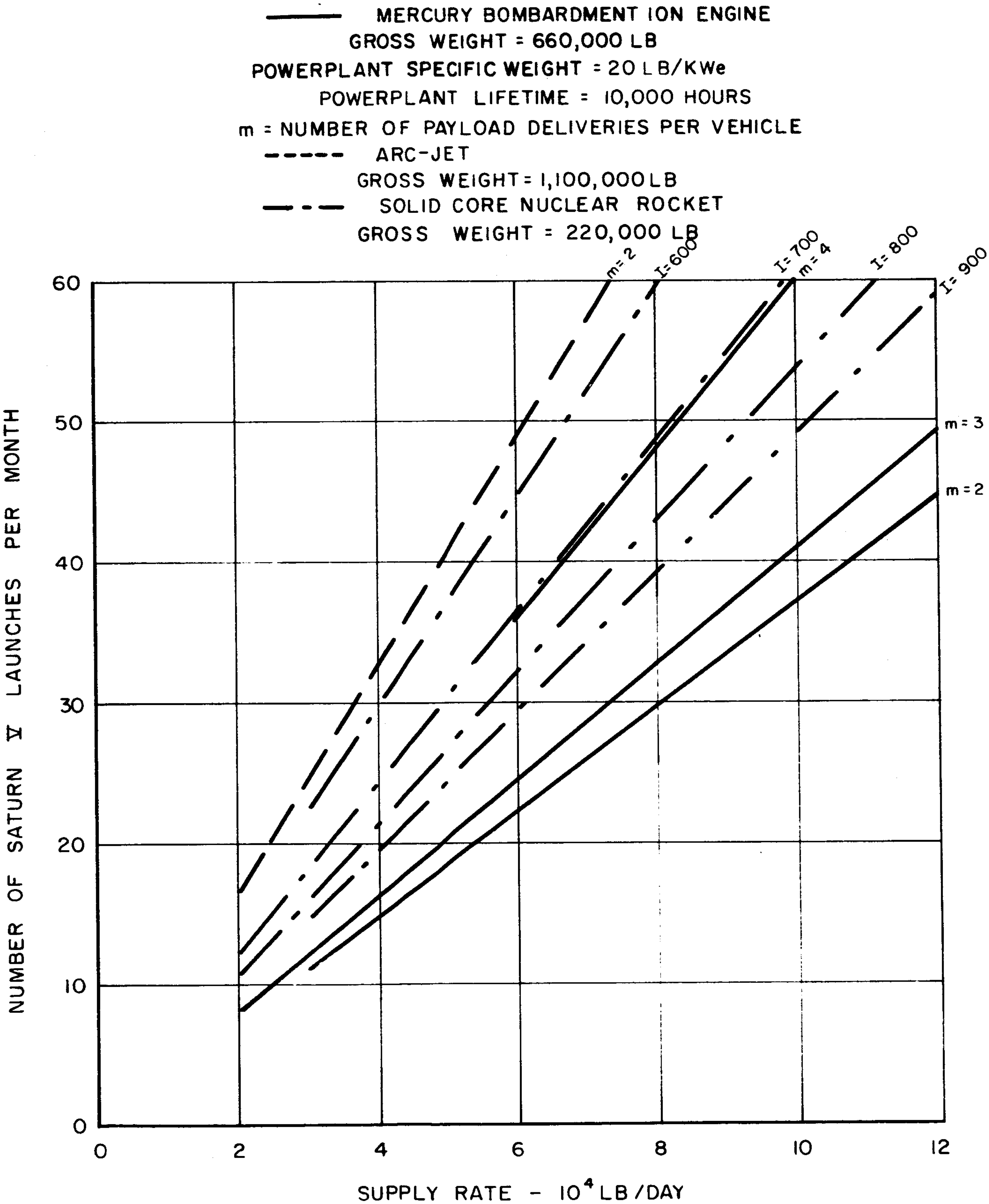
# LUNAR LOGISTIC SUPPLY OPERATION

SPECIFIC SUPPLY RATE VERSUS POWERPLANT FRACTION
HYDROGEN FUELED ARC-JET
POWERPLANT LIFETIME = 10,000 HOURS
POWERPLANT SPECIFIC WEIGHT = 20 LB/KWe
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE



# LUNAR LOGISTIC SUPPLY OPERATION

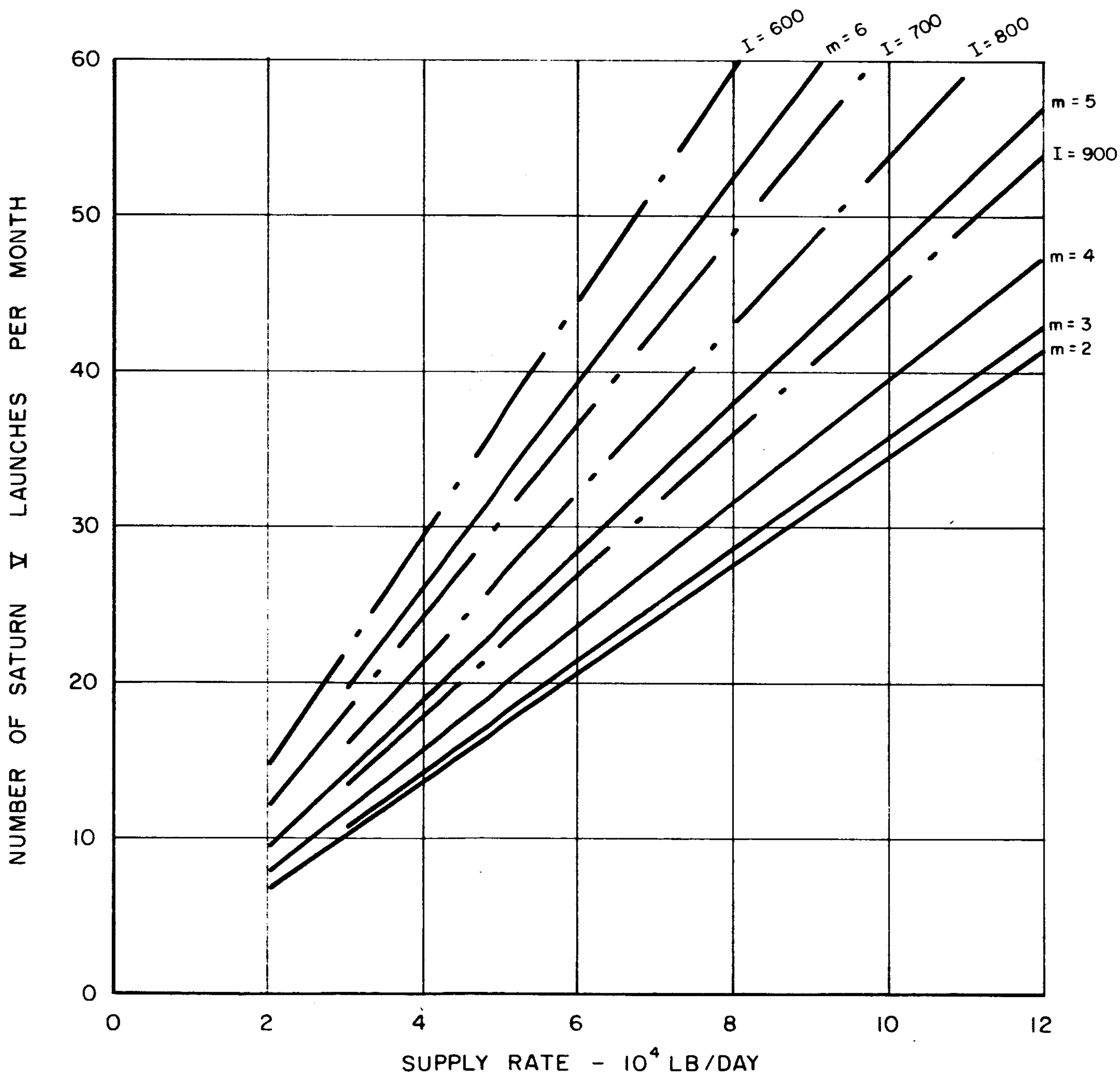
SATURN V LAUNCH RATE



# LUNAR LOGISTIC SUPPLY OPERATION

SATURN V LAUNCH RATE

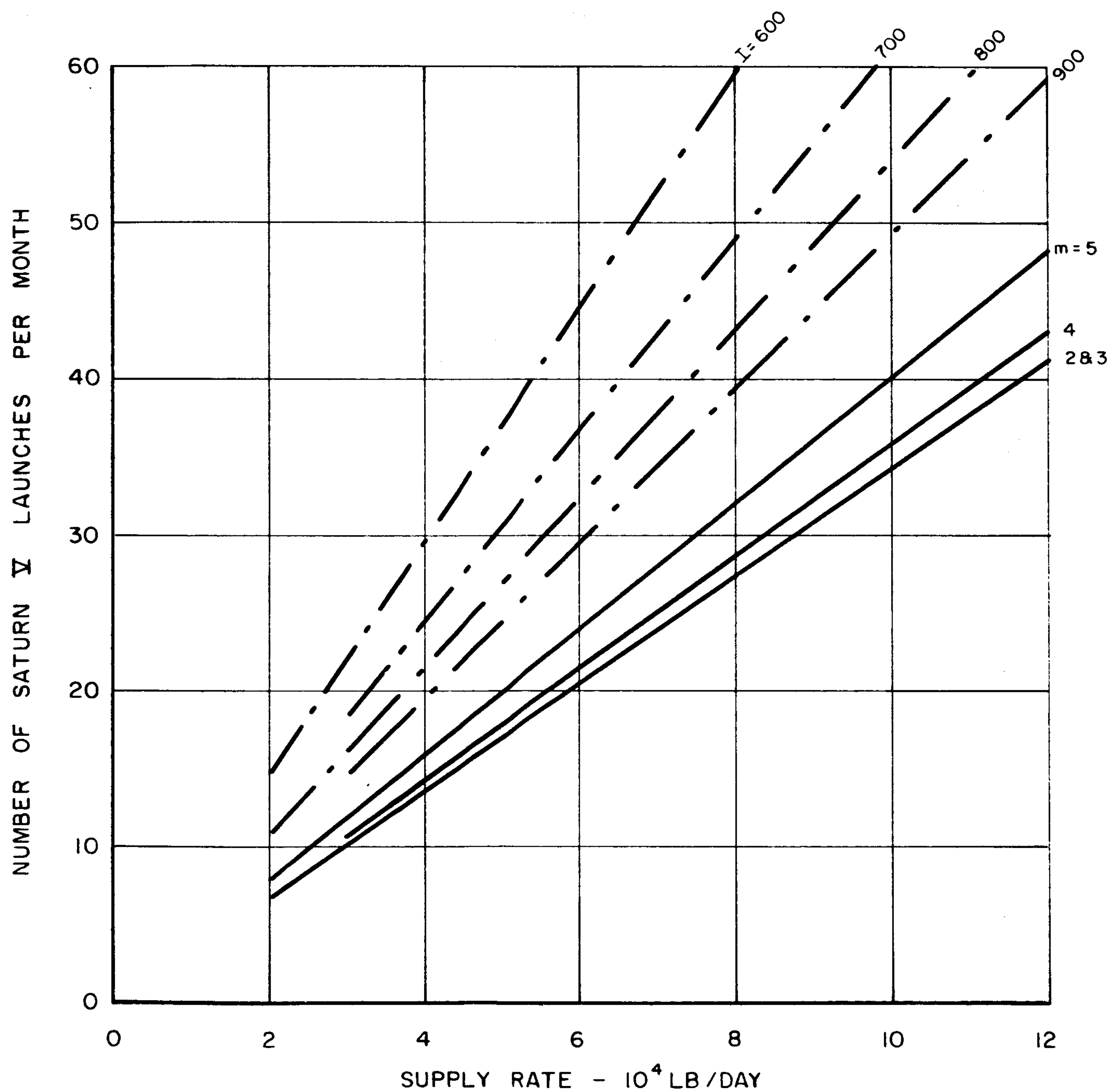
——— MERCURY BOMBARDMENT ION ENGINE
GROSS WEIGHT = 660,000 LB
POWERPLANT SPECIFIC WEIGHT = 20 LB/KWe
POWERPLANT LIFETIME = 15,000 HOURS
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE
— — — SOLID CORE NUCLEAR ROCKET
GROSS WEIGHT = 220,000 LB



# LUNAR LOGISTIC SUPPLY OPERATION

SATURN V LAUNCH RATE

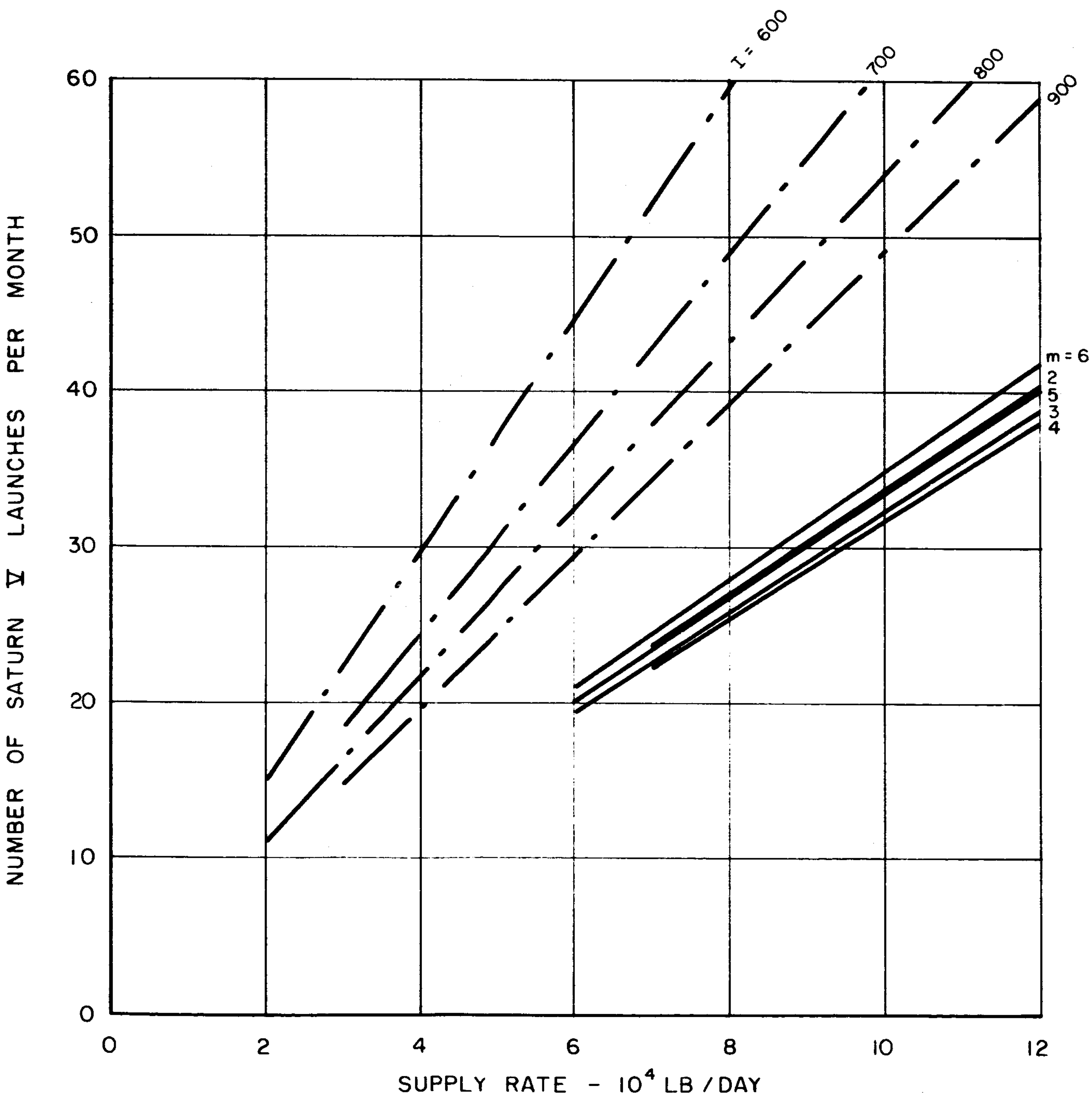
——— MERCURY BOMBARDMENT ION ENGINE
GROSS WEIGHT = 660,000 LB
POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
POWERPLANT LIFETIME = 10,000 HOURS
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE
— - — SOLID CORE NUCLEAR ROCKET
GROSS WEIGHT = 220,000 LB



# LUNAR LOGISTIC SUPPLY OPERATION

SATURN V LAUNCH RATE

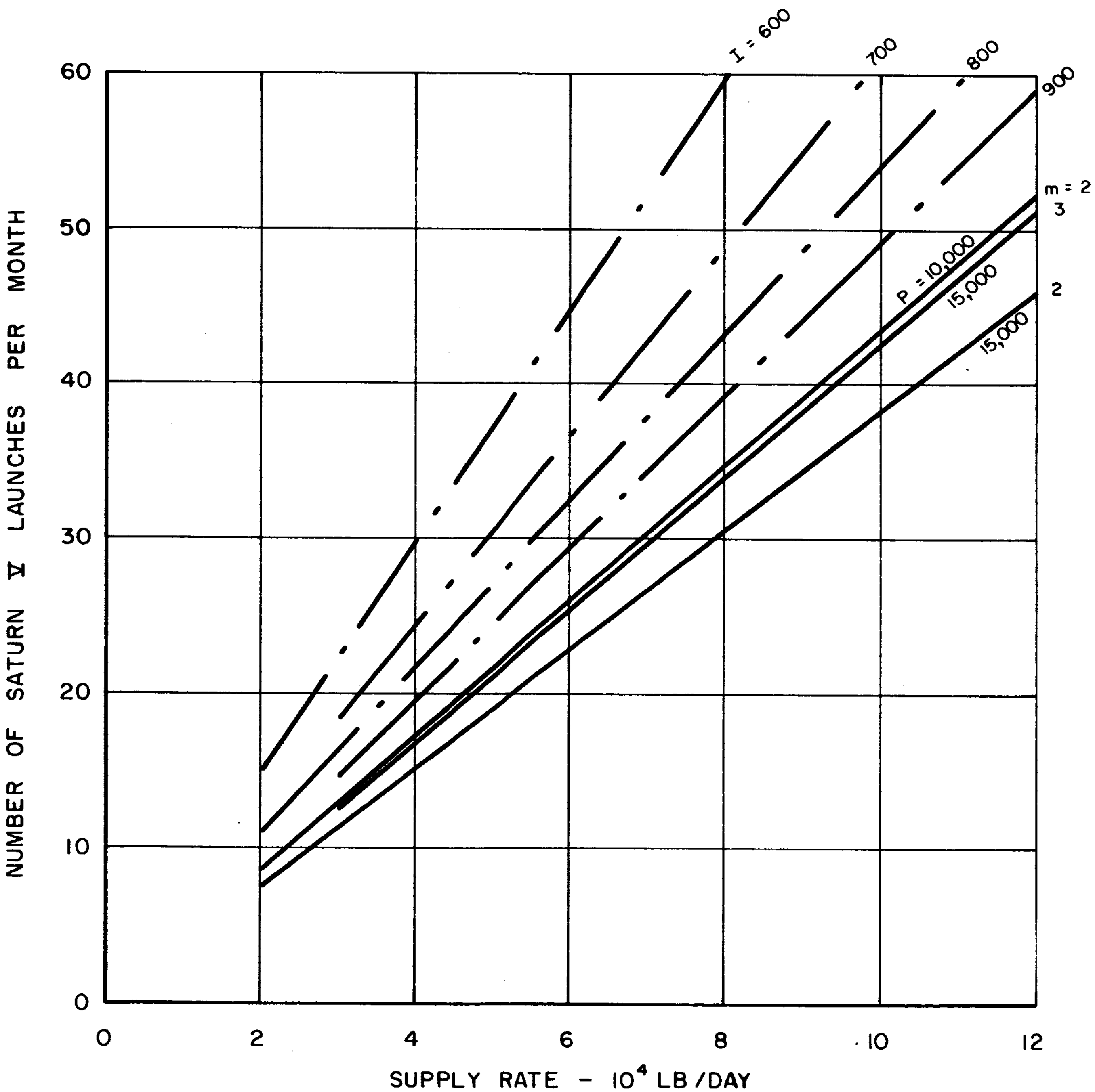
——— MERCURY BOMBARDMENT ION ENGINE
GROSS WEIGHT = 660,000 LB
POWERPLANT SPECIFIC WEIGHT = 10 LB/KWe
POWERPLANT LIFETIME = 15,000 HOURS
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE
— — — SOLID CORE NUCLEAR ROCKET
GROSS WEIGHT = 220,000 LB

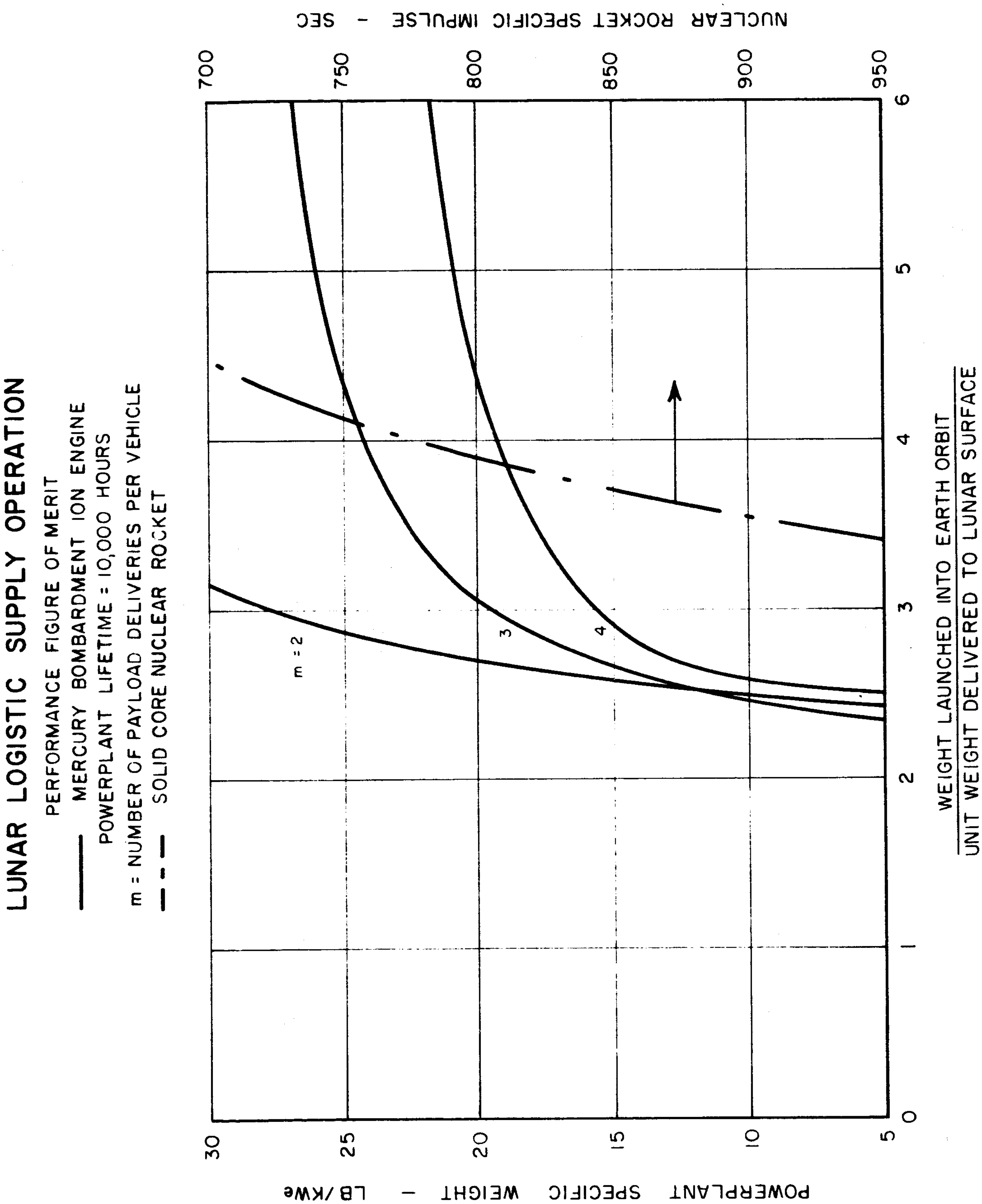


# LUNAR LOGISTIC SUPPLY OPERATION

SATURN V LAUNCH RATE

——— MERCURY BOMBARDMENT ION ENGINE
GROSS WEIGHT = 660,000 LB.
POWERPLANT SPECIFIC WEIGHT = 30 LB/KWe
P = POWERPLANT LIFETIME IN HOURS
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE
— — — SOLID CORE NUCLEAR ROCKET
GROSS WEIGHT = 220,000 LB




LUNAR LOGISTIC SUPPLY OPERATION
PERFORMANCE FIGURE OF MERIT
MERCURY BOMBARDMENT ION ENGINE
POWERPLANT LIFETIME = 10,000 HOURS
m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE
SOLID CORE NUCLEAR ROCKET
NUCLEAR ROCKET SPECIFIC IMPULSE - SEC
700
750
800
850
900
950
POWERPLANT SPECIFIC WEIGHT - LB/KWe
30
25
20
15
10
5
0
1
2
3
4
5
6
m = 2
3
4
WEIGHT LAUNCHED INTO EARTH ORBIT
UNIT WEIGHT DELIVERED TO LUNAR SURFACE

## LUNAR LOGISTIC SUPPLY OPERATION

PERFORMANCE FIGURE OF MERIT

—— MERCURY BOMBARDMENT ION ENGINE

POWERPLANT LIFETIME = 15,000 HOURS

m = NUMBER OF PAYLOAD DELIVERIES PER VEHICLE

— - — SOLID CORE NUCLEAR ROCKET